Near Earth Asteroid Returned Sample

MARTIN MARIETTA
ASTRO SPACE

LOWELL OBSERVATORY

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A Discovery Mission Concept

Principal Investigator
Eugene M. Shoemaker
Lowell Observatory
Flagstaff, AZ 86001
602-556-7181

Spacecraft and Technical Lead
Andrew F. Cheng
The Johns HopkinsUniversity
Applied Physics Laboratory
Laurel, MD 20723-6099
301-953-5415

Consortium Partners
Lowell Observatory
Applied Physics Laboratory
Martin Marietta Astrospace

Co-Investigators

T. J. Ahrens	California Institute of Technology
R. P. Binzel	Massachusetts Institute of Technology
W. V. Boynton	University of Arizona
D. E. Brownlee	University of Washington
D. S. Burnett	California Institute of Technology
A. F. Cheng	The Johns Hopkins University Applied Physics Laboratory
R. Farquhar	The Johns Hopkins University Applied Physics Laboratory
M. J. Gaffey	Rensselaer Polytechnic Institute
J. Jones	NASA Johnson Space Center
I. Lewis	Lawrence Livermore National Laboratory
J. McAdams	Johns Hopkins Applied Physics Laboratory
L. McFadden	University of Maryland
H. McSween	University of Tennessee
S. Ostro	Jet Propulsion Laboratory
T. Swindle	University of Arizona

JHU/APL Study Team Members

A. F. Cheng	Study Manager
D. W. Dunham	Mission Design
R. W. Farquhar	Mission Design
E. H. Frye	Presentation Coordinator
A. D. Goldfinger	Asteroid Operations
P. J. Grunberger	Mission Operations
D. Y. Kusnierkiewicz	Spacecraft System Engineer
J. V. McAdams	Mission Design
B. A. Northrop	Report Documentation
E. L. Reynolds	Spacecraft Systems
S. R. Vernon	Sample Collector

Martin Marietta Aerospace Study Team

R. Brewer	Return Capsule Lead Engineer
D. McGaughey	Return Capsule Mechanical Engineer

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- **NASA Discovery Program:**
  - Development cost less than $150 M (FY92 dollars)
- **Mission Concept:**
  - Return to Earth 10-100 g from each of four to six sites on a near Earth asteroid
  - Perform global characterization of the asteroid and measure mass, volume and density to 10%
- **Consortium Partners:**
  - JHU/APL: provide spacecraft (NEAR derivative) and integration, provide sample collection system, perform mission operations
  - Lowell Observatory: Eugene Shoemaker, Principal Investigator
  - Martin Marietta Astrospace (Valley Forge): provide Earth return capsule
- **Launch Opportunities:**
  - Prime mission is January 2000 launch to (4660) Nereus, probably a primitive C-type asteroid; backup mission is January 2002 launch to Nereus
- **Launch Vehicle:**
  - Delta II-7925

Overview

Mission Concept:

Return to Earth 10-100 g from each of four to six sites on a near-Earth asteroid. Perform global characterization of the asteroid and measure mass, volume and density to 10%.

Objectives:

- Provide first direct and detailed petrological, chemical, age, and isotopic characterization of a near-Earth asteroid and relate it to terrestrial, lunar, and meteoritic materials.
- Sample the asteroid regolith and characterize any exotic fragments.
- Identify heterogeneity in the asteroid's isotopic properties, age, and elemental chemistry.

Target Asteroids:

- 4660 Nereus, probably a primitive C-type asteroid (prime target)
- 1989ML, an extremely accessible asteroid of unknown type (alternate target)

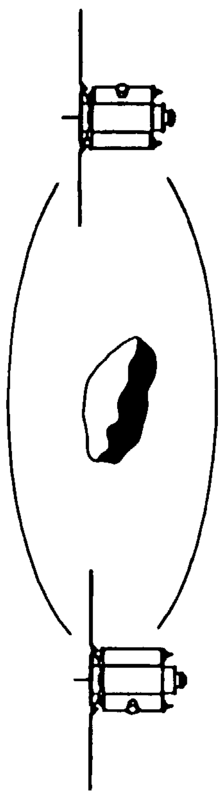
Launch Vehicle: Delta II-7925

Overview *(Cont.)*

- Accommodate only instruments required for landing and sample collection
  - Determine asteroid shape, density, gravity field, rotation state
  - Obtain moderate resolution images of surface including sample sites
- Collect 4 to 6 samples of mass >10 gm each and return to Earth
- Autonomous landing and sample collection from rock or regolith surface in near-zero gravity
- Touch-and-go sampling - No long duration landing; spacecraft makes momentary contact with asteroid and obtains sample with pyrotechnic device that fires sampling tube into surface
- Pyrotechnic sampling device is currently under development at APL and has successfully obtained > 40 gm samples from hardened concrete and loose sand
- Warm sample return - Asteroidal rock and regolith samples can be handled like lunar samples with existing techniques and facilities

Mission Overview

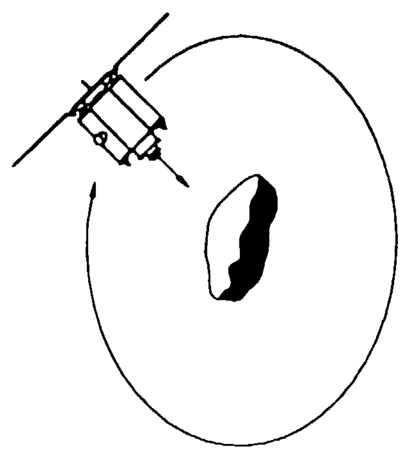
1. Orbital Survey Phase



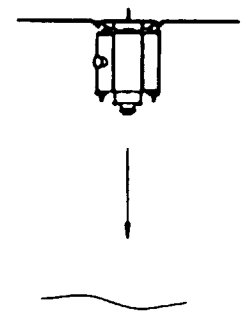
Point high gain antenna to Earth, instruments to asteroid, sun within 70° of full lumination.

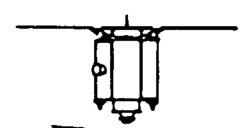
2. Landing Phase

2a. Deorbit Burn



2b. Quasi vertical descent; point fan beam to Earth, sampler to asteroid, and sun within 70° of full illumination.

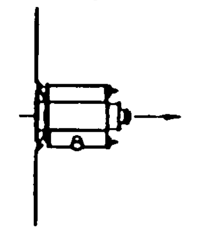




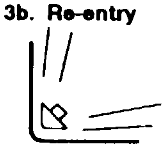
2c. Touch and go sampling; no long duration landing; no robotic manipulator arm; obtain total of six samples.

3. Earth Return

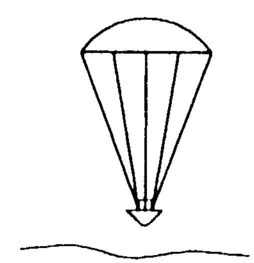
3a. Separation



3b. Re-entry



3c. Land Recovery



(4660) Nereus

Type:	C? (definitely not Type S)
Perihelion:	0.953 AU
Aphelion:	2.03 AU
Inclination:	1.43°
H Magnitude:	18.3
Size:	1-km class
Rotation:	Unknown

Survey of Missions: 1998-2004

Launch Date	Target Asteroid	Return Date	Total ΔV (km/s)
Jan 1998	Nereus	Feb 2002	5.79
Jan 2000	Nereus	Feb 2004	5.62
Jan 2002	Nereus	Feb 2006	5.54
Jul 2004	Nereus	Feb 2011	5.21

Strawman Payload

Instrument	Heritage
Survey camera with filters capable of lithologic discrimination	BMDO - Clementine flight spare unit
Descent Imager	Clementine flight spare
Laser Altimeter	NEAR LIDAR
Sample Collector	
Earth Return Capsule	Pioneer Venus Probes
Radio Science	Many missions

Spacecraft Summary

- Based on NEAR design
- Solar powered, three-axis stabilized, dual mode propulsion
- Maximum wet mass 792 kg
- Maximum dry mass 473 kg, including
  - Instruments 9 kg
  - Sampling system 21 kg
  - Return capsule 27 kg
- Spacecraft ΔV capability 1475 m/s
- X-band telemetry

Modifications to NEAR Design

Payload:

- Replace 55 kg NEAR payload with 4 kg for survey and descent imagers
- Add six-shooter (sample collector) mounted in re-entry capsule, no manipulator arm
- Modify NEAR LIDAR for short range operation

Spacecraft:

- No change to power system
- No change to RF system except fanbeam antenna location
- minor changes to propulsion system
- No hardware changes to command and telemetry, guidance and control, attitude subsystems

Summary

- NEARS will lead to breakthroughs in our understanding of planetesimal formation and evolution
- NEARS will put asteroid science and meteoritics in a new regime
- NEARS is a pathfinder for future robotic sample return missions

Science

- NEARS will obtain the first detailed history of a celestial body beyond Earth and Moon
- NEARS will be a decisive test of linkage between meteoritics and asteroid science and will lead to breakthroughs in both fields, just as Apollo samples produced breakthroughs in lunar science
- By returning samples for laboratory analyses, NEARS will establish formation conditions, formation ages, and gross dynamical history
- Measurements will be made that are not feasible with remote sensing, such as determination of isotopic reservoirs, crystallization and shock ages, cosmic ray exposure ages, thermal evolution, and collisional evolution

Science Objectives

- Provide the first direct and detailed petrological, chemical, age, and isotopic characterization of a near-Earth asteroid
- Relate the asteroid's petrology, elemental abundances, and isotopic ratios to laboratory results for terrestrial, meteoritic, and lunar materials
- Sample the asteroid regolith, and if possible, measure its depth and characterize its stratigraphy
- Identify any significant heterogeneity in the body's isotopic properties, age, and elemental chemistry over micro to global scales
- Characterize exotic fragments not typical of the bulk asteroid

Science Requirements

- Sample four to six sites, with individual samples of 10 to 100 grams
- Maintain samples below 400K during reentry and recovery
- No requirement to maintain vacuum seal
- Obtain global characterization of asteroid's gross compositional structure
- Measure asteroid's rotation state, shape, mass, and density (to about 10%)

Strawman Payload

Survey Imager:	**250 μr resolution** **75 x 100 mr FOV** **6-position filter wheel**
Descent Imager:	**250 μr resolution** **75 x 100 mr FOV** **6-position filter wheel**
Laser Altimeter:	**Acquisition 20 km** **range resolution 2 m**
Sample Collector:	**Six samples, >10 gm each,** **rock or regolith surface**
Radio Science:	**Two-way Doppler to 0.1 mm/s**

Touch-and-Go-Sampling

- No long duration landing
- Spacecraft makes momentary contact with asteroid and obtains sample with pyrotechnic device that fires sampling tube into surface
- Sample can be obtained in zero gravity
- Sample can be obtained from rock or regolith surface
- Pyrotechnic sampling device is under development at APL and has operated successfully into concrete and sand targets
- Accommodate a cluster of sampling tubes (“six-shooter”) for multiple samples

Scientific Rationale

- State-of-the-art analysis techniques can be applied to returned samples, even as the state of the art evolves.
- The experimental protocol can be more flexible, with further experiments designed on the basis of previous, unexpected results.
- Some critical analyses that require heavy sample preparation (e.g., geochronology) are simply not possible for an automated system.
- Proves the sampling technology for more challenging sample return missions (e.g., a comet nucleus).

Specific Justification for a Near Earth Asteroid Return Sample Mission

- Provides an understanding of the relationships between two massive databases (asteroid spectrophotometry and meteorite petrology).
- If above item is accomplished, can bootstrap into the main asteroid belt (the source for near Earth asteroids), and there address fundamental questions about the formation of the solar system.
- Provides insights into regolith evolution and space weathering processes on small bodies.
- Provides extraterrestrial materials of high chemical pristinity.
- Allows the possibility of obtaining samples with geologic context, so as to unravel the geology of asteroidal bodies.

Mission Design Status

- **Baseline Mission Overview**
  - January 2000 launch to (4660) Nereus
  - Trip time = 49 months, Stay time = 70 days
  - Ten-day launch window requires ΔV = 1.475 km/s on board
- **Baseline Comparison With Current NEAR Mission**
  - Maximum solar range nearly identical (2.2 AU)
  - On-board propulsion budget at 1.500 km/s for NEAR
  - Much closer to Sun (1.2 to 1.0 AU) while at Nereus
  - Much closer to Earth (0.3 to 0.1 AU) while at Nereus
  - Launch energy slightly higher for NEARS

Survey of Missions: 1998-2004

Except for the 1998 launch opportunity, all NEARS mission options on the facing page may be accomplished with minor NEAR propulsion system modification. These missions met conservative design requirements for using the NEAR propulsion tanks modified to hold 10% more bipropellant. The 1998 launch is most likely too early, and the higher launch energy indicates a need for significant reduction in dry spacecraft mass. The numbers shown in the table correspond to the minimum total ΔV solution (except for the 2002 launch case which corresponds to center of launch window). Therefore, the stay time and post-launch ΔV will vary slightly throughout the launch window. The baseline mission is highlighted in a box. All other known asteroid sample return missions in this time period failed to meet both the launch energy and post-launch ΔV requirements for launch aboard the Delta II-7925 using the basic NEAR spacecraft design. Other important constraints include a minimum 60-day stay time and Earth return velocity less than 7 km/sec. These constraints are chosen to allow sufficient time for asteroid characterization and sample collection (stay time) and to minimize heat shield mass for the Sample Return Capsule.

Survey of Missions: 1998-2004

Launch Date	Target Asteroid	Stay Time (days)	Return Date	Post-Launch ΔV (km/s)	Total ΔV (km/s)
Jan 1998	Nereus	262	Feb 2002	1.235	5.788
Jan 2000	Nereus	68	Feb 2004	1.265	5.619
Jan 2002	Nereus	87	Feb 2006	1.286	5.543
Jan 2004	Nereus	1434	Feb 2011	1.211	5.209

Baseline mission is highlighted in box.

2000 Launch Nereus 4-Year Sample Return

The trajectory profile as viewed from the north ecliptic pole appears on the facing page for a launch date late in the launch window. The launch energy (C_3) varies from 25.8 to 26.3 km^2/s^2 across the 10-day launch window. After nearly one revolution of the Sun, rendezvous with Nereus occurs in October, 2001. The ΔV for rendezvous varies from 766 to 708 m/s from open to close of the launch window. Asteroid characterization and sampling operations occur during the Nereus stay time, which ranges from 70.2 to 67.5 days from open to close of the launch window. Nereus departure and Earth return dates, Nereus departure ΔV, and relative velocity at Earth return all remain constant across the launch window. This 49-month mission ends with a return trajectory having slightly more than one revolution of the Sun starting with 715 m/s ΔV at Nereus and ending with a return capsule velocity of 6.553 km/s relative to Earth. This Earth-relative velocity corresponds to a 12.850 km/s entry velocity. As with NEAR, the NEARS spacecraft has about 2.2 AU as its maximum solar range.

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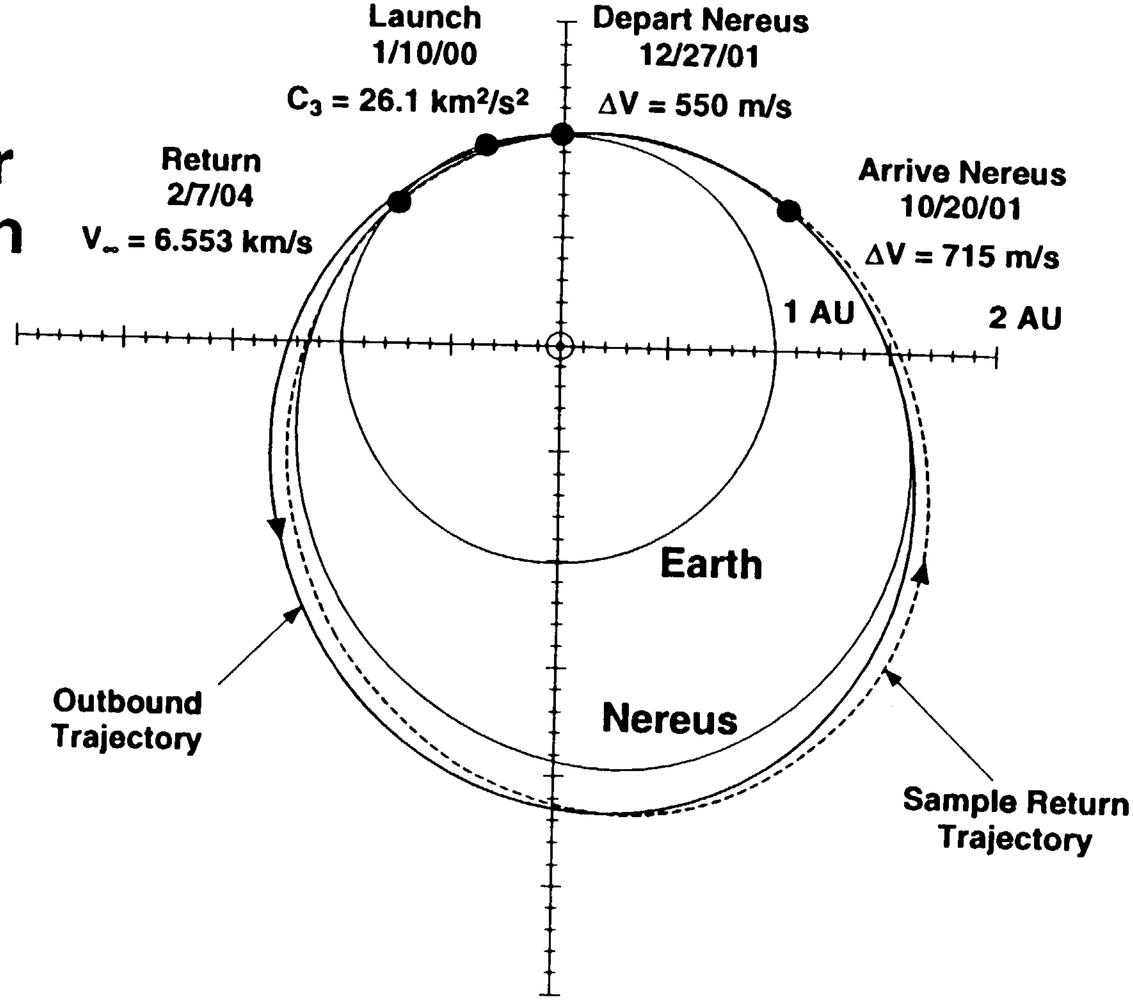
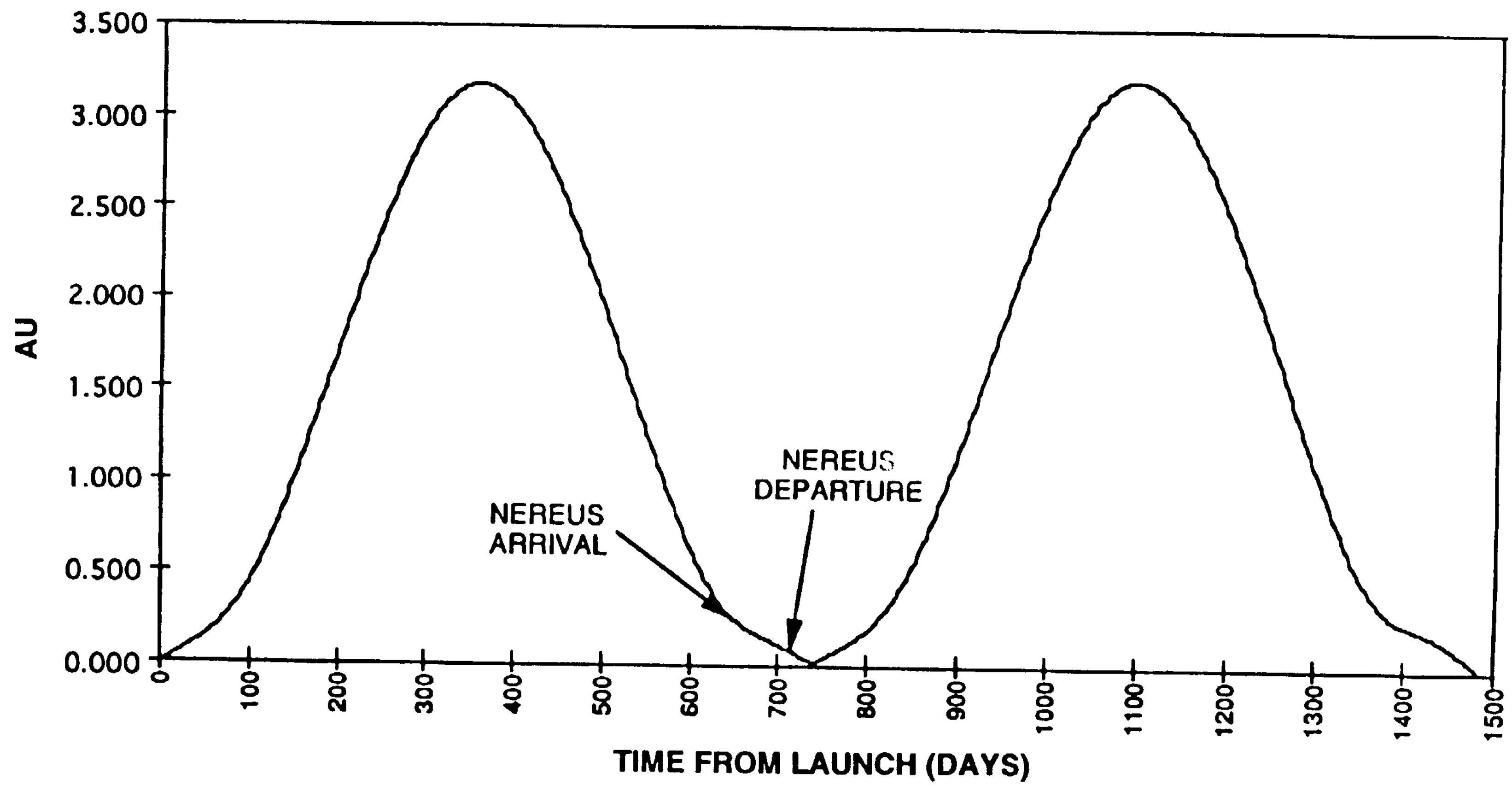
2000 Launch Nereus 4-Year Sample Return



Spacecraft-Earth Distance

The NEARS spacecraft-Earth distance approaches a mission-long minimum during and shortly after the time the spacecraft is at Nereus. This geometry results in a range of two-way communications times from 1.5 to 4.5 minutes during the rendezvous and departure propulsive maneuvers as well as during the closely monitored sample collection phase. The close proximity of Earth also allows downlink of data through the fanbeam antenna during the latter portion of operations at Nereus, when the spacecraft's high-gain antenna will not be pointed toward Earth. During this phase of the mission, the solar panels will be pointed within 70° of full illumination to supply adequate solar power. Shortly after Nereus departure (when the spacecraft comes within 0.01 AU of Earth), both the low-gain and the fanbeam antennas can be used to downlink data.

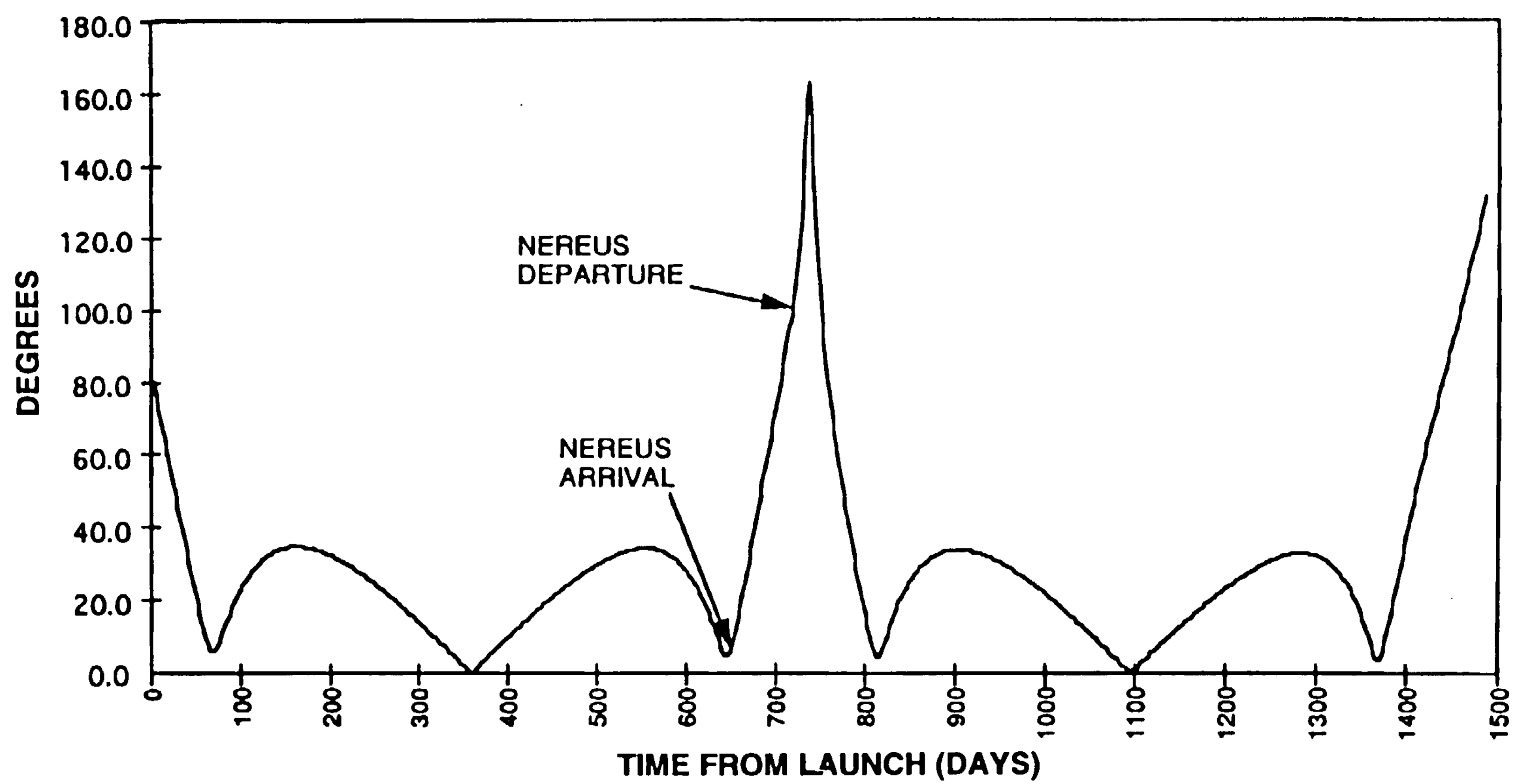
S/C - Earth Distance



Phase Angle (S-V-E)

The NEARS Sun-vehicle-Earth (S-V-E) angle approaches a mission-long maximum during and shortly after the time the spacecraft is at Nereus. While the high-gain antenna points toward Earth the available power is approximately equal to the cosine of the phase angle times the power available at zero-phase angle (solar panels perpendicular to the Sun-spacecraft direction). Note that, except for the near-Earth portions of the mission, the phase angle remains less than 37°. This represents no more than a 20% decrease from the power at full illumination during the cruise phase of the mission. During the sampling phase at Nereus the S-V-E angle exceeds 60°, so the solar panels will be oriented within 70° of full illumination and will generate adequate power because Nereus is near 1 AU from the sun. During this time, Earth is also within 0.15 AU of Nereus, allowing downlink of data through the fanbeam antenna at an adequate data rate.

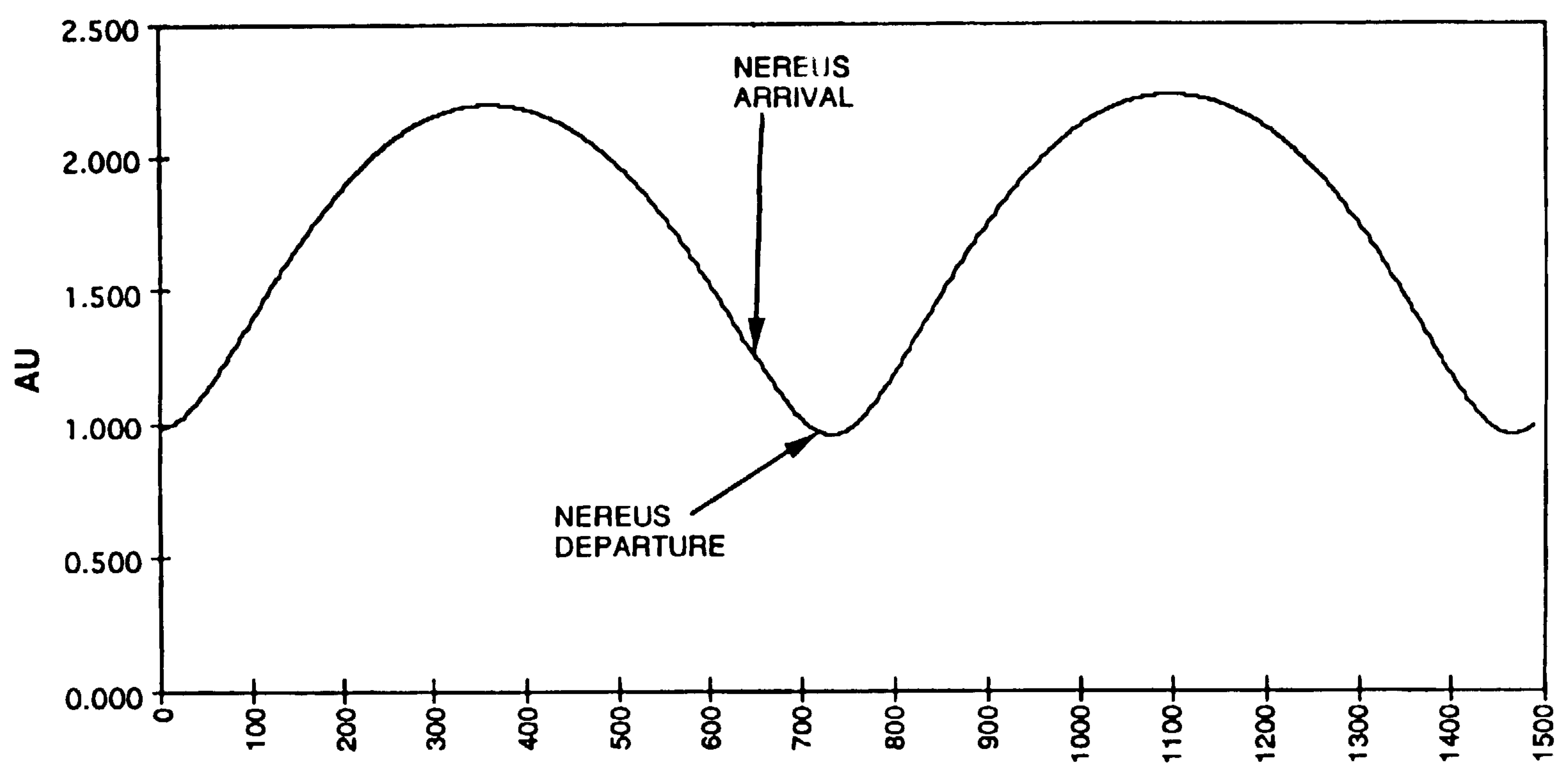
Phase Angle (S-V-E) (Degrees)



Spacecraft-Sun Distance

The NEARS spacecraft aphelion (maximum distance from the Sun) of 2.24 AU is almost identical to the aphelion for the NEAR spacecraft. One key difference is that for NEAR a propulsive maneuver occurs near aphelion where the solar panels supply about 1/5 the power they supply at 1 AU from the Sun. However, for NEARS the spacecraft is in a low power requirement cruise mode at or near aphelion. The maximum solar range for a planned NEARS propulsive maneuver is less than 1.3 AU where three times the power level at aphelion is available (assuming full-Sun orientation for the solar panels). Throughout the time at or near asteroid Nereus this solar range decreases to less than 1 AU.

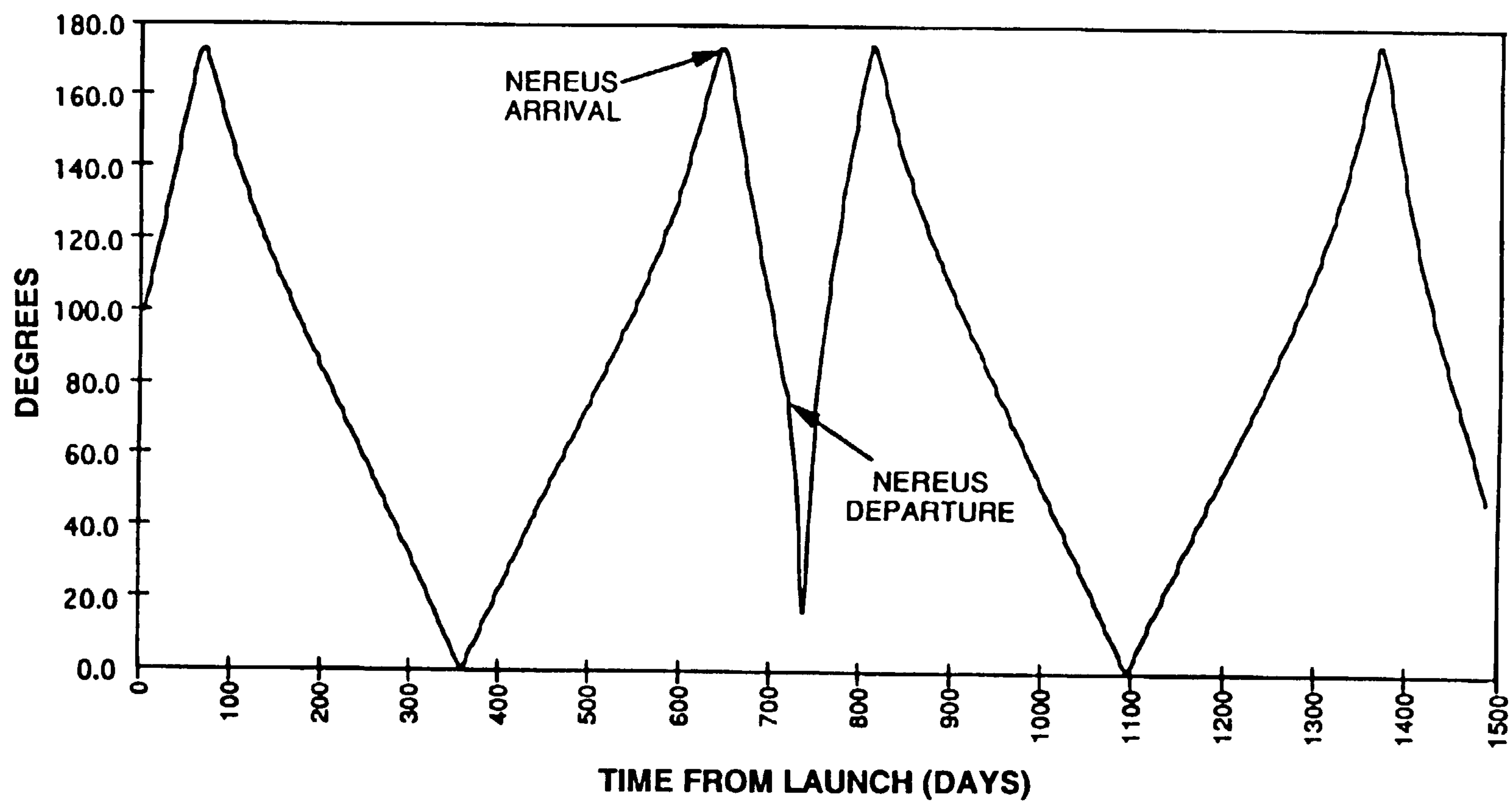
S/C - Sun Distance



Solar Elongation Angle (V-E-S)

The NEARS solar elongation or vehicle-Earth-Sun (V-E-S) angle must be greater than 2° during the time from Nereus arrival to Nereus departure in order to avoid unacceptable solar interference with uplink transmissions. From Nereus arrival to Nereus departure the solar elongation angle remains above 75°, well above the constraint. The plot on the facing page represents the value of solar elongation angle for day 8 in the 10-day launch window. The margin of about 75° beyond the constraint value ensures that, throughout the launch window and for post-departure correction maneuvers, no interruption in command transmission to the spacecraft will occur. The final design will ensure that no communications blackout will occur at a critical mission phase. At two other times in the mission the solar elongation angle constraint is approached or violated. However, this is not a problem since these incidents are during cruise phase, when communication with the spacecraft is intermittent.

Solar Elongation Angle (V-E-S)



Baseline Mission Launch Window

Launch window design for the baseline mission indicates availability of a 10-day period during which a 792 kg spacecraft can be injected by the Delta II-7925 upper stage from a 185 km parking orbit into a heliocentric transfer trajectory to Nereus. The 8-foot diameter payload fairing enshrouds the NEARS spacecraft during much of the launch ascent phase. The launch window open/close dates are determined by the launch energy corresponding to the NEARS required spacecraft mass. For example, at the highest launch energy (26.276 km^2/s^2) in the launch window the Delta II-7925 with 8-foot fairing can inject 792 kg payload mass. The 10-day launch window opens at about 11 AM on January 2, 2000.

No mass penalty or insufficient post-launch ΔV is incurred for this launch window due to high DLA (declination of launch asymptote). The -9.5° DLA is well below the 28.5° no-penalty limit. The lowest post-launch ΔV margin occurs at the opening of the launch window. Here the 1475 m/s ΔV is apportioned as follows:

initial injection error correction	50 m/s
Nereus rendezvous	766 m/s
Nereus rendezvous 2% adjustment	15 m/s
Nereus orbital and sampling	25 m/s
Nereus departure	550 m/s
Nereus departure 2% adjustment	11 m/s
navigation (incl. Earth return targeting)	58 m/s

The highest post-launch ΔV margin (116 m/s for navigation) occurs at the close of the launch window. Stay time at Nereus varies from 70.2 days to 67.5 days from open to close of the launch window.

Baseline Mission Launch Window

Launch Date	Day In Window	Launch Energy* (km^2/s^2)	Launch Asymptote Declination (deg)	Post-Launch Deterministic**	ΔV(m/s) Nav/Correction
Jan 2.5, 2000	1	26.245	-16.3	1316	159
Jan 7.0, 2000	5.5	25.796	-14.7	1287	188
Jan 11.5, 2000	10	26.276	-13.2	1258	217

Notes: * Spacecraft mass = 792 kg (payload mass at highest launch energy in launch window delivered by Delta II-7925 with 8-foot diameter payload fairing at 95% probability of command shutdown).

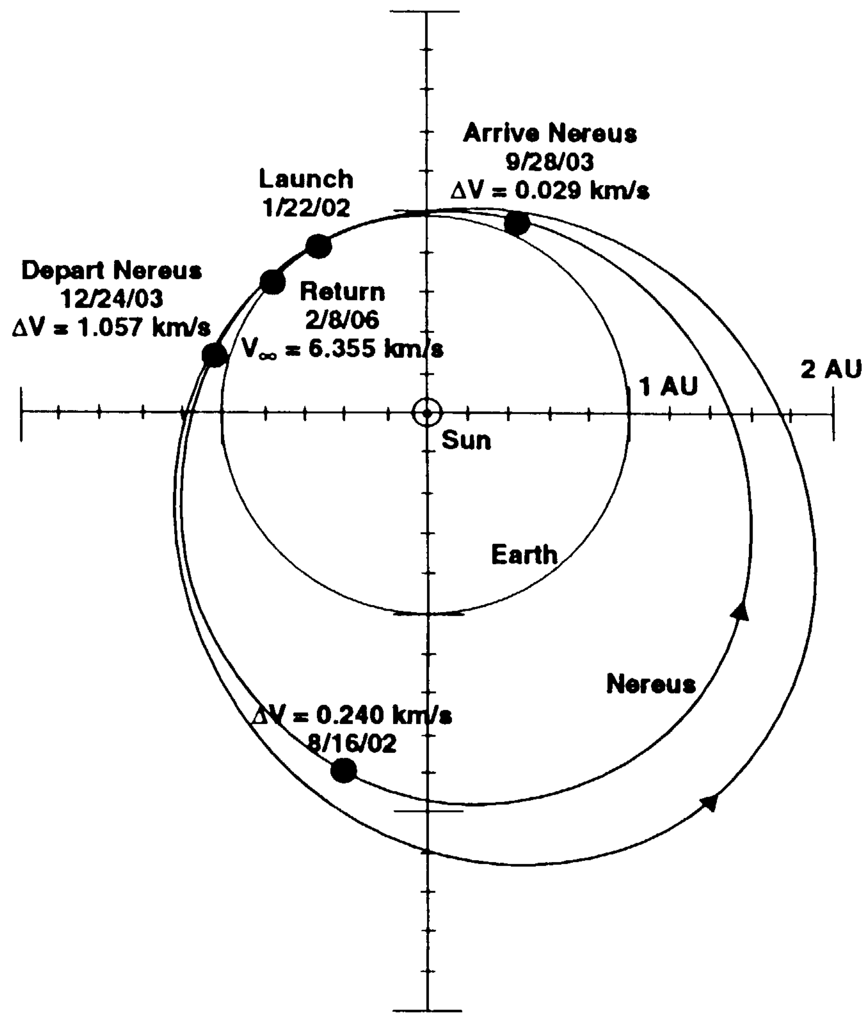
** Deterministic ΔV includes Nereus rendezvous and departure ΔVs.

Backup Mission - 2002 Launch Nereus 4-Year Sample Return

The trajectory profile as viewed from the north ecliptic pole appears on the facing page for a launch date early in the launch window. Preliminary analysis revealed existence of at least a 10-day launch window throughout which launch energy and post-launch ΔV requirements are met. Due to the small ΔV at Nereus arrival, much of the trajectory from the deep space maneuver (240 m/s) to Nereus arrival lies very close to the orbit of Nereus. Therefore, at the scale used for this plot, the spacecraft trajectory and Nereus' orbit are indistinguishable. Another advantage to this backup mission choice is that the relative velocity at Earth return is slightly less than for the baseline mission. This requires no design changes for the Earth return capsule aeroshell.

Backup Mission -

2002 Launch Nereus 4-Year Sample Return



Mission Design Conclusions

- Interplanetary trajectory similar to NEAR (design heritage)
  - Launch energy (spacecraft bus, launch vehicle)
  - 4-year mission (component design lifetime)
  - Lower Earth distance at Nereus (telecommunications)
  - Minimum/maximum solar distance (thermal/power)
  - Post-launch ΔV budget (propulsion tanks)
  - Asteroid rendezvous (orbit insertion scheme, orbital operations)
- Nereus 2002 backup mission can be accomplished with identical spacecraft design

Imaging System

- NEARS imaging system includes two small CCD cameras: a survey imager and a descent imager
- Survey imager to be used for shape and rotation state determinations as well as large scale mapping of asteroid
- Survey imager is a flight spare Clementine UV-Visible imager without modification
- Descent imager to be used for close-up imaging of sample sites and characterization of their geologic context
- Descent imager is another flight spare Clementine UV-Visible imager, with filter wheel modified to accommodate close-up lenses

Clementine UV/Visible Camera

- **Miniature and modularized ultraviolet (visible) imager with intermediate field of view**
  - Phosphor-overcoated charge coupled detector (CCD) for UV and visible response
  - Filter wheel selects spectral bands of interest; camera optimized for sun-illuminated objects
  - 20 camera images and 5 filter wheel positions per second
  - Ground surveillance with 100 meter resolution from low Earth orbit (existing optics)
- **Current on-orbit UV-Visible sensors are 2x - 4x heavier, require 3x more power and are expensive**
  - Designed for inexpensive manufacture and calibration; optics, filter wheel and camera assembly are modular
  - Common control and data bus architecture for ease of integration and test

Clementine UV/Visible Imager

Focal Length: 90 mm

288 x 384 pixels; 23 micron pixels

Field of view: 4.2 x 5.6 degrees

Spectral range: 0.28 to 1.0 microns

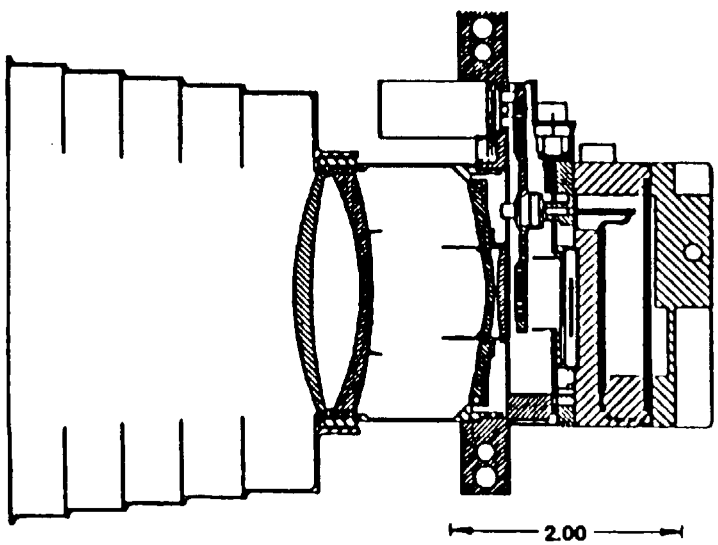
Six position filter wheel

Frame rate: 20 Hz

Mass: 500 gm

Size (cm): 10.5 x 12 x 16

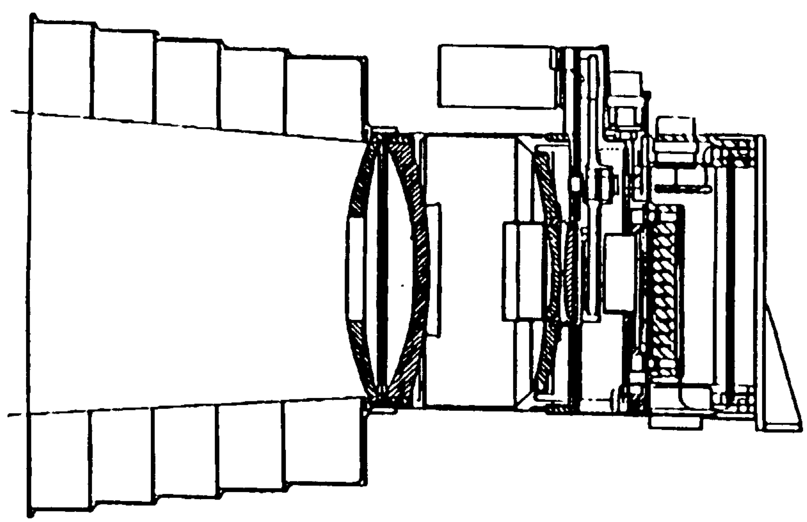
Clementine UV/Visible Sensor



Physical Parameters

- 6.1 W CCD, electronics, stepper motor hold power
- Conductively cooled/heated
- Integral mounting features
- Rotating filter wheel
  - Size 5 stepper motor, 90° step angle, > 2000 hr life
  - rim gear wheel
  - 6:1 gear ratio
  - <±5 mrad repeatability
  - two radial bearings
  - <200 msec step + settle time
  - 12.0 W step power
  - 1.2 W hold power
- Baffle/lens mounted to camera housing
- Mass <483 g

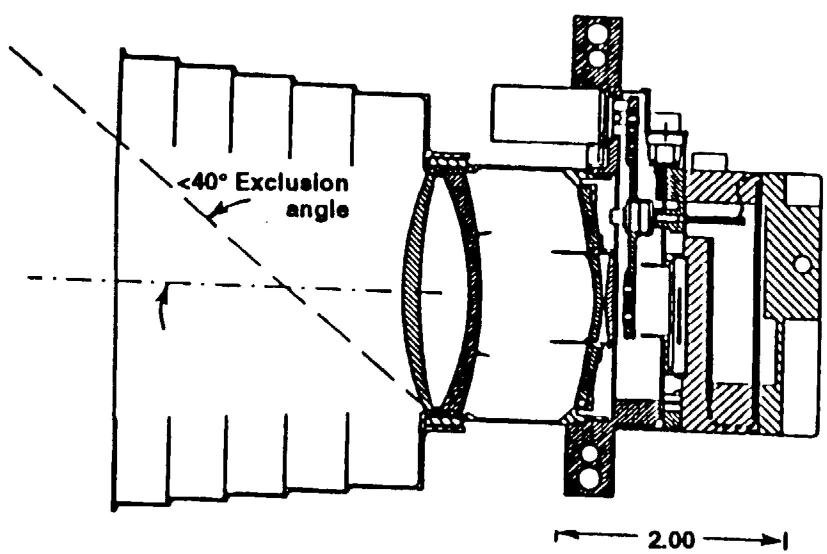
UV/Visible Sensor



Electronic Parameters

Focal plane array	TH 7863CRH-UV-01-B/T Metachrome II Q.E. >8% at 250 nm
A/D resolution	8 bits
Frame rate	30 Hz
Digitization	150 e^- / cnt 350 e^- cnt 1000 e^- / cnt
Readout noise	<60 RMS e^-
Integration control	13 bits LSB = 94.4 Ms
Power	
– camera	4.5 W
– filter wheel	1.2 W hold 12.3 step

UV/Visible Sensor



Design Features

Entrance pupil diameter	46 mm
Effective focal length	90 mm
Spectral band	415 ± 20 (nm)
	750 ± 5
	900 ± 10
	950 ± 15
	1000 ± 15
	400 to 950
Format	6.624 x 8.832 mm
Image quality	30 μm φ
Transmission	>20% to >50%
Distortion	<1%

Stray Light Reduction Features

Internal and external baffle
Black anodized housing

Reentry Capsule Requirements

- Protect and return asteroid samples for land recovery, possibly White Sands
- Direct entry velocity 12.85 km/s
- Entry path angle determined considering thermal protection, parachute deployment and targeting accuracy issues
- Sample weight, 200 gm
- Sample container weight 15-20 kg
- Container size, 25.4 cm diameter x 30.5 cm length
- Sample to be maintained below 400°K

Design Approach

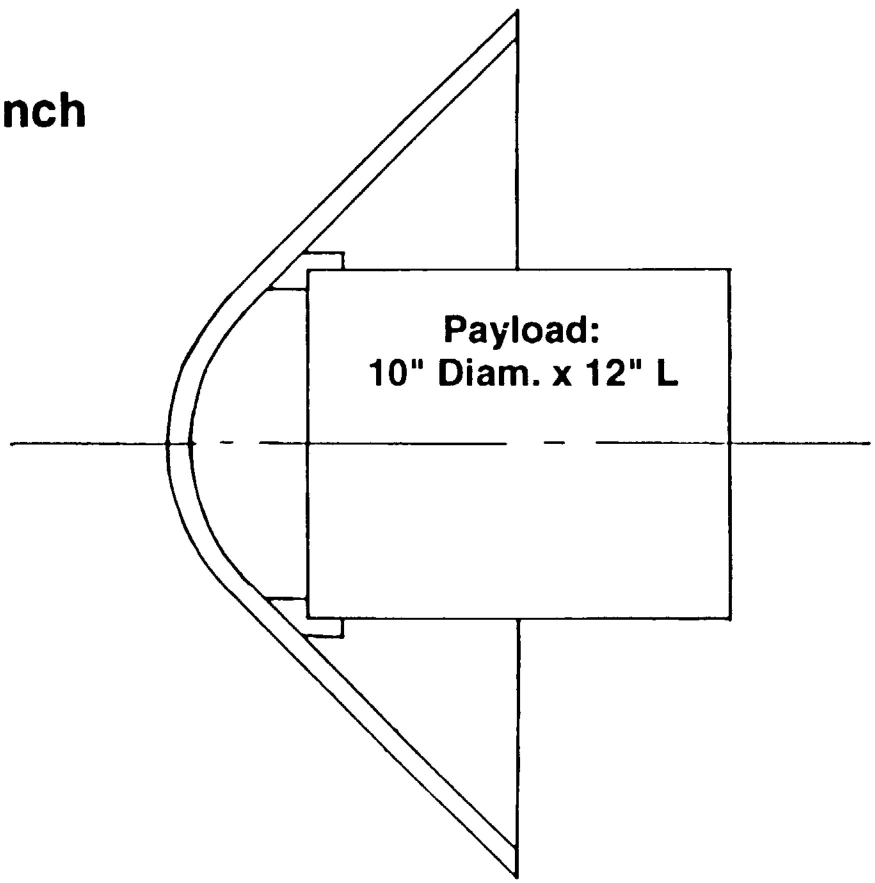
- Utilize Pioneer Venus heritage
- Employ scaled-down PV-small probe aeroshell (4/5 scale)
- Payoffs in aerothermodynamic design, i.e., extensive data available that can be utilized in NEARS program
  - Wind tunnel data for aero characteristics
  - Ablation data on CMCP (Nosecap) and TWCP (Frustum)
- Thermal environment for direct return entry similar to Venus entry - thus, heat shield/structure design changes are minimal
- Currently, subsystem weights for various subsystems, i.e., parachute, power, electronics and pyros, are scaled from PV designs
- Mechanical design of sample collector needs to be integrated into return capsule design
  - Attachments, door closure, sealing, temperature control, etc.

Return Capsule Configuration

- Aerodynamic design uses Pioneer Venus heritage
- D_{BASE} is the base diameter
- R_{BASE} is the base radius
- R_N is the nose radius

Return Capsule Configuration

- D_{BASE} = 25 Inch
- R_B/R_N = 2.0

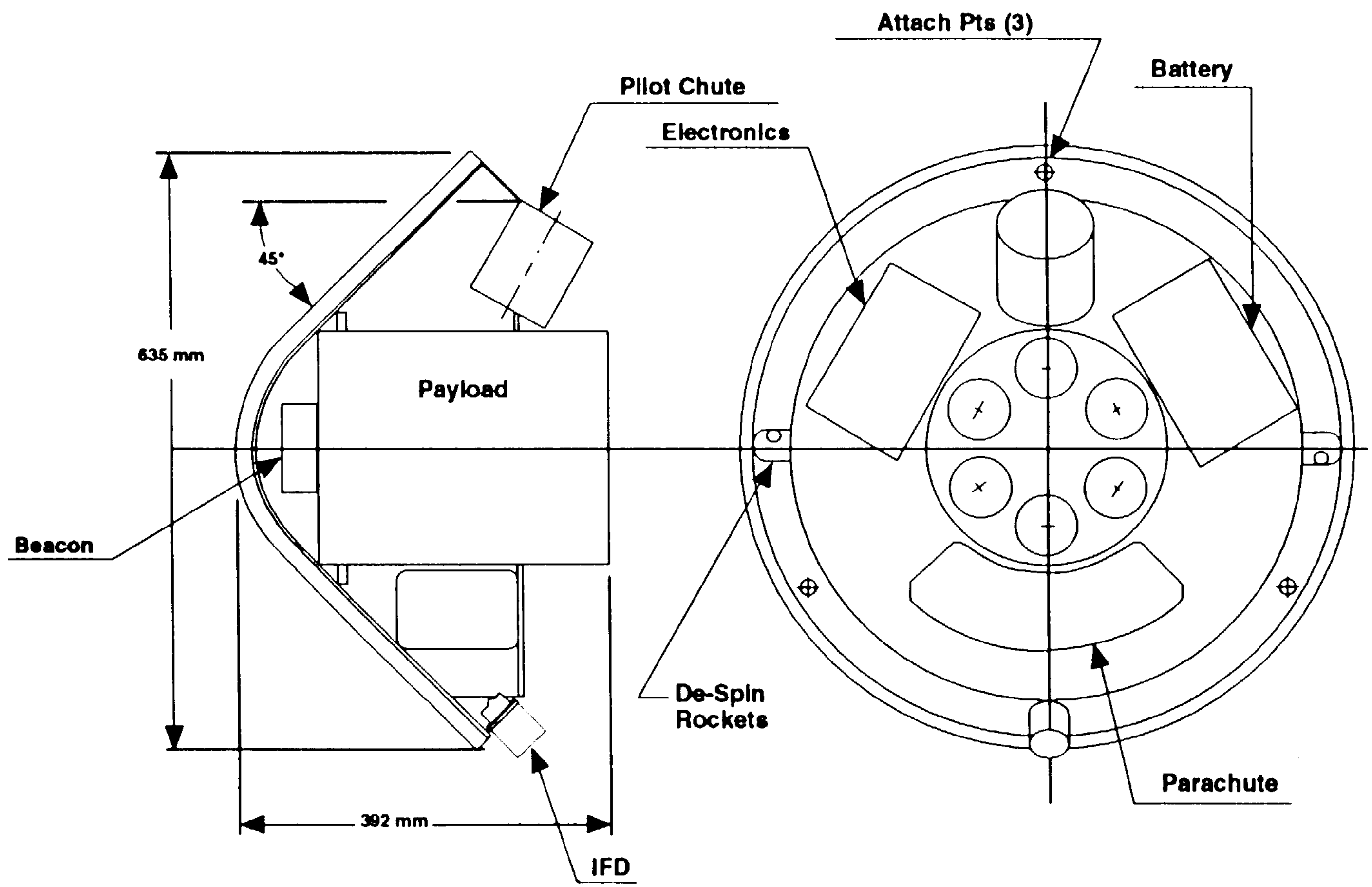


Near Earth Asteroid Returned Sample

MARTIN MARIETTA
ASTRO SPACE

LOWELL OBSERVATORY

Reentry Capsule

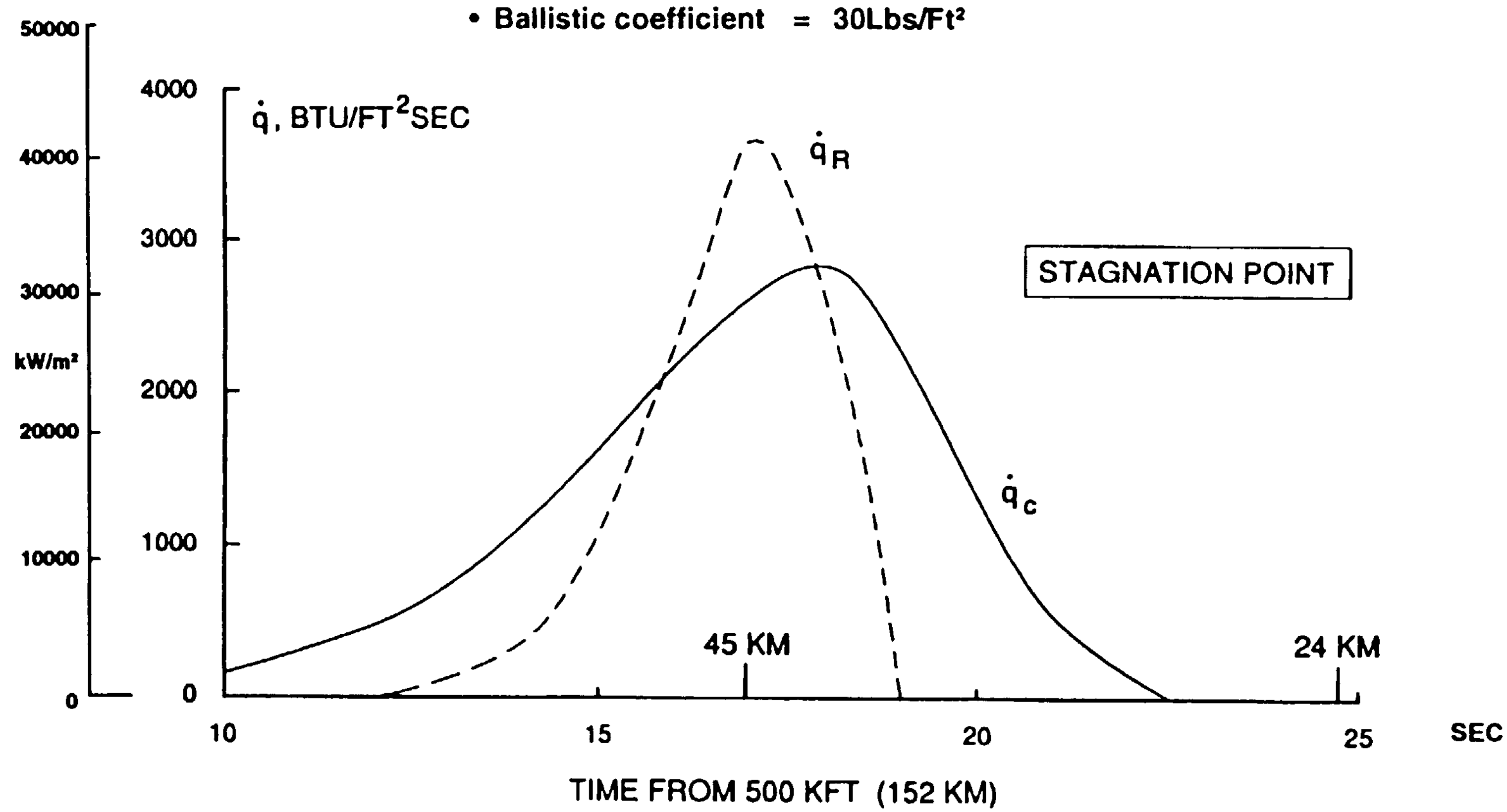


Thermal Protection Requirements

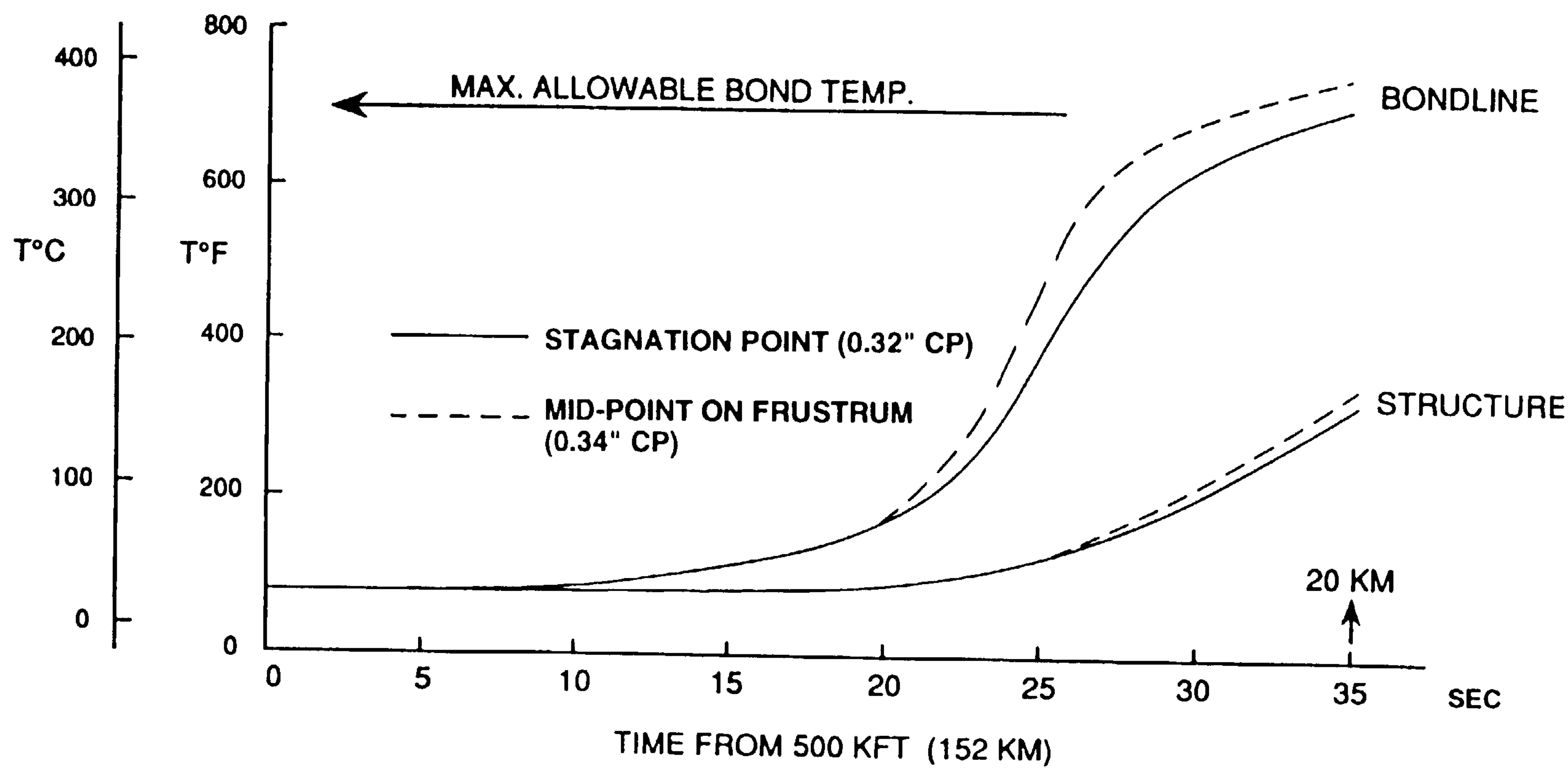
- Pioneer Venus design
- Nosecap - Chopped Molded Carbon Phenolic (CMCP)
- Frustrum - Tape Wrapped Carbon Phenolic (TWCP)
- Base Area - Low Density Elastometric Shield Material (ESM)
- Thermal environment calculated using approximate analysis and tailored to PV experience
- Heat shield thermal response predicted using 1-D Rekap Code (Martin Marietta Astro Space Computer Program)
  - Predicts recession, degradation and temperature profiles
- Heat shield bonded to 0.05 inch titanium structure with 0.040 inch RTV 630

Capsule Entry Thermal Environment

- Entry velocity = 12.85 km/sec
- Entry angle = 30°
- Ballistic coefficient = 30Lbs/Ft²



Predicted Temperature Time Profiles



Summary of Heat Shield Requirements

- Radiative heating calculated using Tauber & Sutton, J. Spacecraft Vol. 28, No. 1, Jan 1991
- Criterion for TPS Sizing: Maximum allowable temperature of 700°F (644°K) on RTV 630 bondline

Location	Nom. Thickness (Inch)	Recession (Inch)	Safety Margin	Design Thickness (Inch)
Stagnation Region	0.32	0.094	1.50	0.48
Mid-Frustrum	0.34	0.154	1.35	0.46

Recommendation: 0.5 inch carbon phenolic over entire forebody

Summary of Return Capsule Mass

Configuration: $D_B = 25$ inch; $R_B/R_N = 2$

Subsystem	Mass ~ Lbs (kg)
Heat Shield	20.6 (9.3)
Structure	17.1 (7.8)
Parachute	9.0 (4.1)
Electronics	4.5 (2.0)
Battery	6.6 (3.0)
Pyros	1.0 (0.5)
Deceleration System	58.8 (26.7)
Payload	46.9 (21.3)
Total	105.7 (48.0)

$$\text{Ballistic Coeff.} = \frac{\text{Mass}}{\text{Drag Coeff.} \times \text{Area}}$$

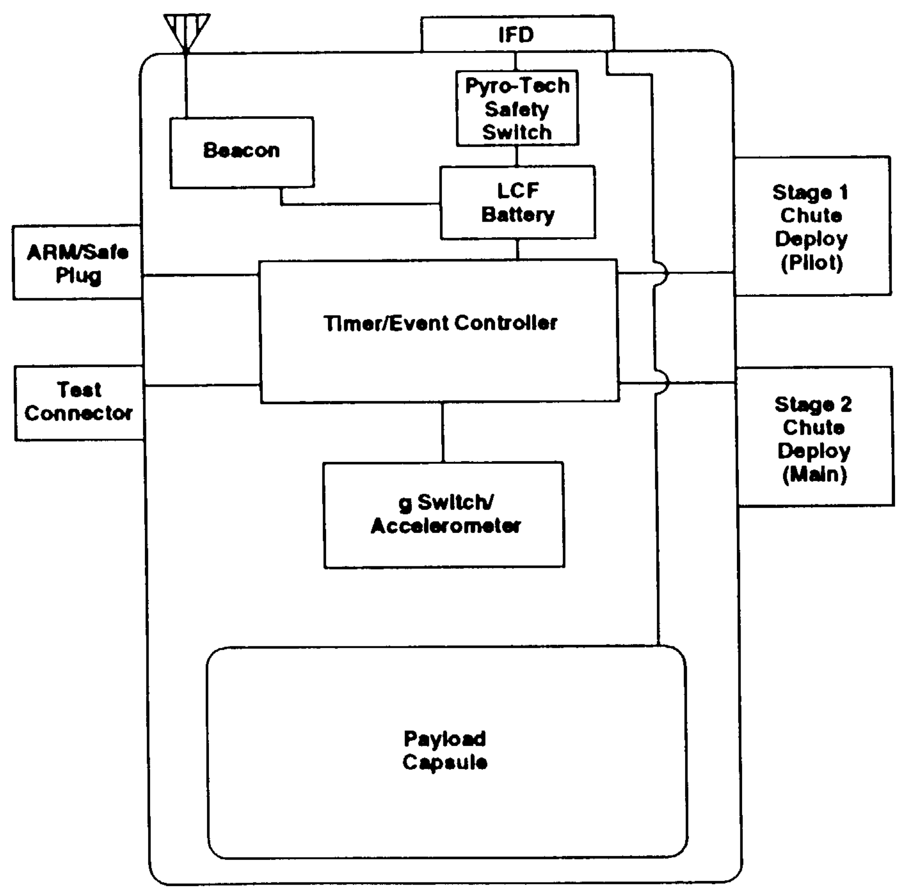
$$\beta = \frac{105.7}{1.05 \times 3.41}\ \text{Lbs/Ft}^2$$

$$\beta = 29.5\ \text{PSF}\ (144\ \text{kg/m}^2)$$

Recovery Vehicle Electrical Block Diagram

- The IFD is the in-flight disconnect.
- The beacon is a UHF transmitter, e.g., Vector T-100 S/L series.
- The battery is lithium carbon monofluorographite.

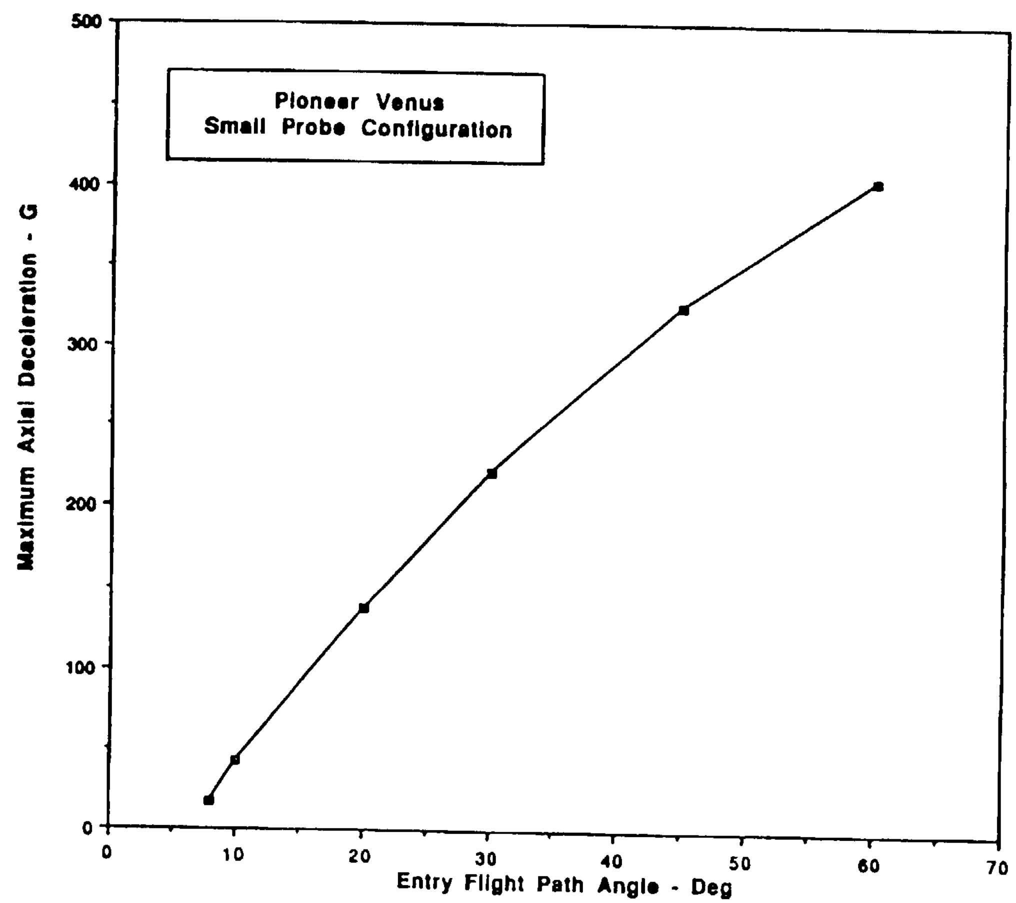
Recovery Vehicle Electrical Block Diagram



Reentry Sequence of Events

(1) Battery activate
(2) Spin up
(3) Deploy
(4) Reentry
(5) g Switch
(6) Pilot chute deploy
(7) Deceleration
(8) Main chute deploy
(9) Reentry continues
(10) Landing/recovery

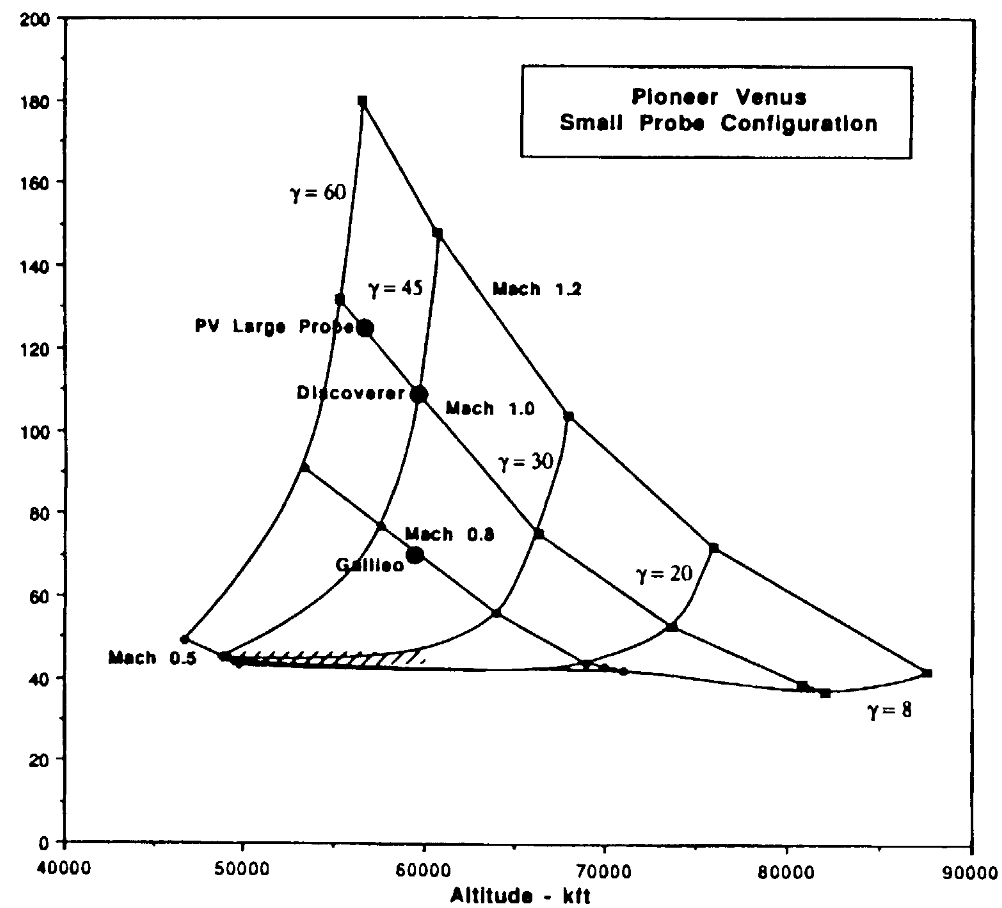
Expected Flight Aerodynamic Loads



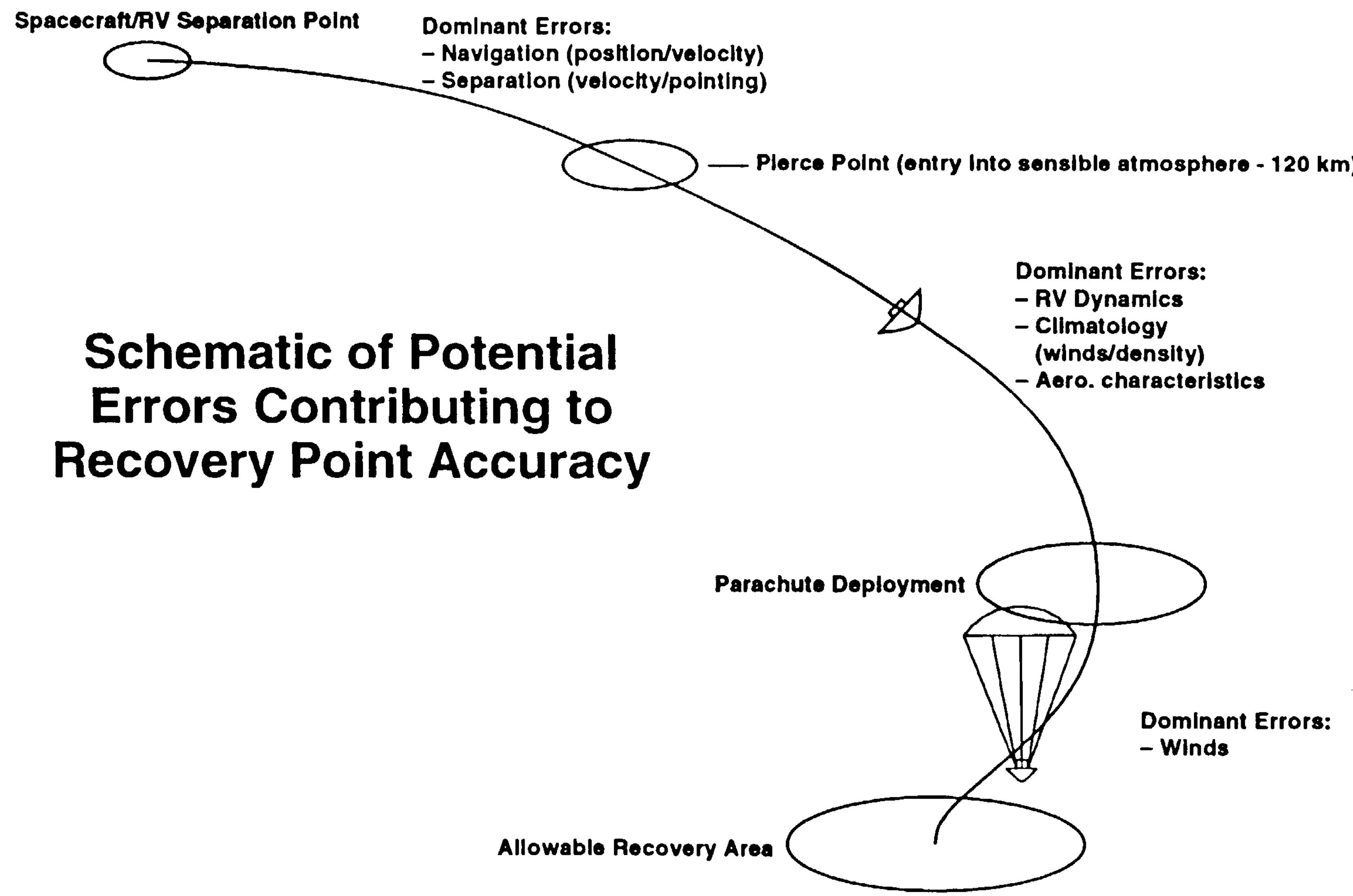
Parachute Deployment Conditions

- Hatch mark area shows desired conditions for NEARS

Parachute Deployment Conditions



Schematic of Potential Errors Contributing to Recovery Point Accuracy

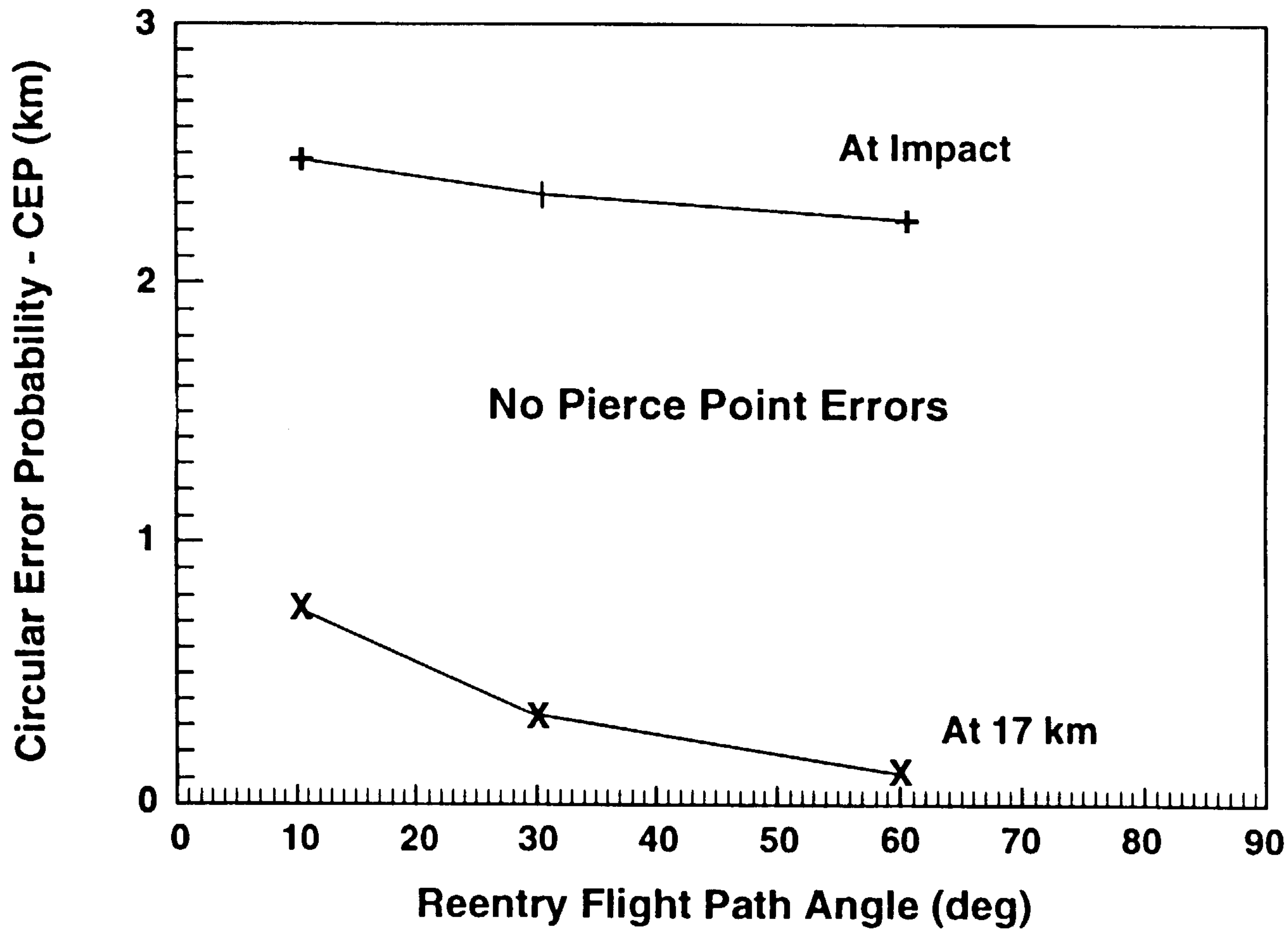


Allocation of Error Sources for White Sands Recovery

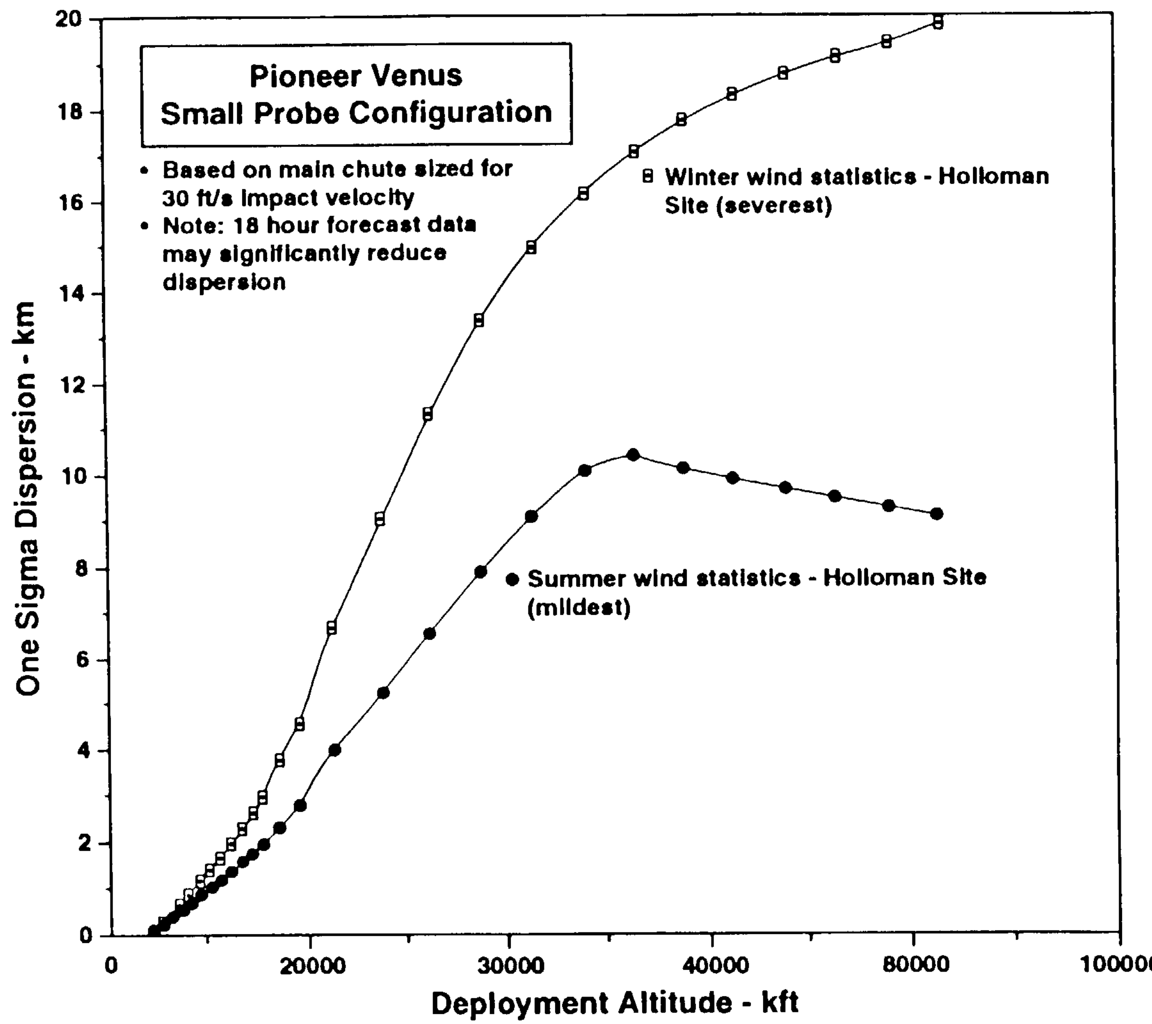
- Assume that allowable 3 sigma miss at White Sands is 16 km
- Assume error sources are independent with total equal to RSS of contributors
- Allocation of error sources (one sigma)

2.1	Reentry (one sigma)
4.0	Parachute drift (one sigma)
2.8	Pierce point - nav/sep (one sigma)
5.3	Total (one sigma) $\Rightarrow$ 16 km (three sigma)

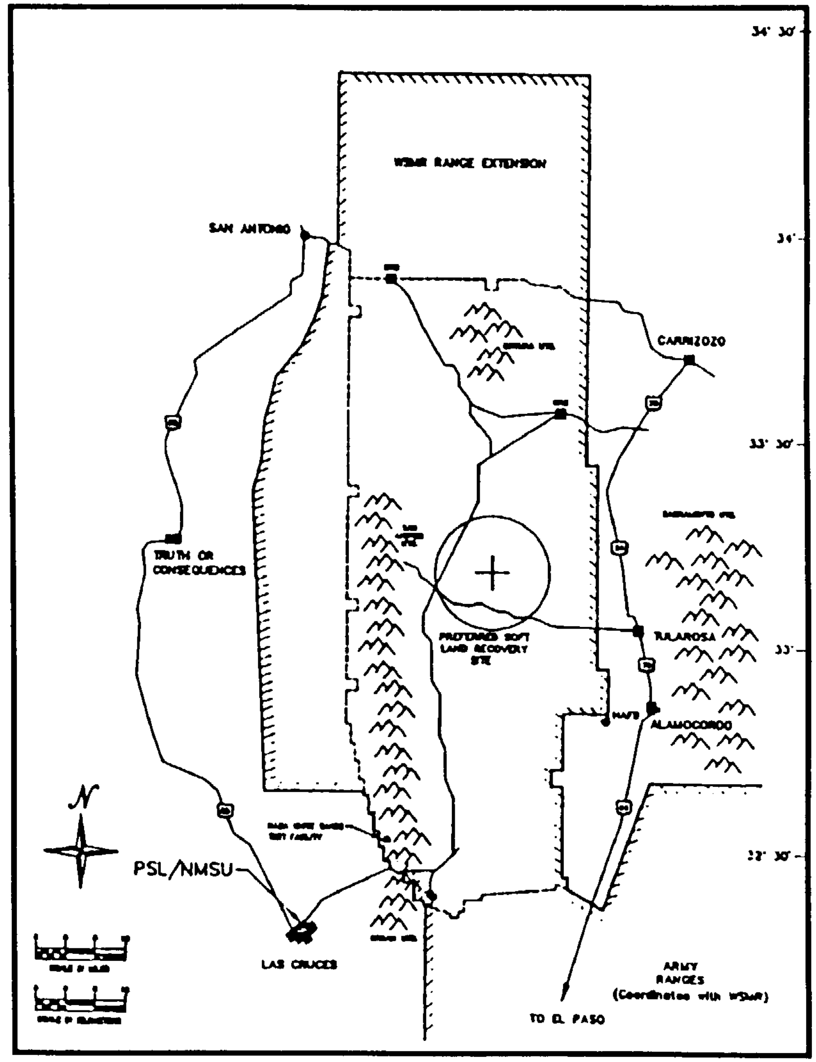
Reentry Accuracy With No Parachute Deployment



Parachute Drift

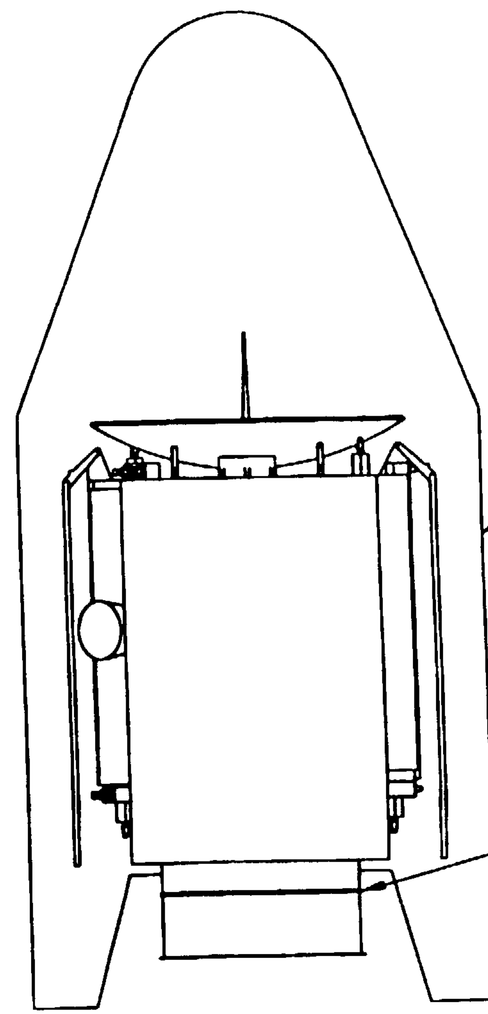


White Sands
Missile Range, NM
Total Effective
Recovery Footprint

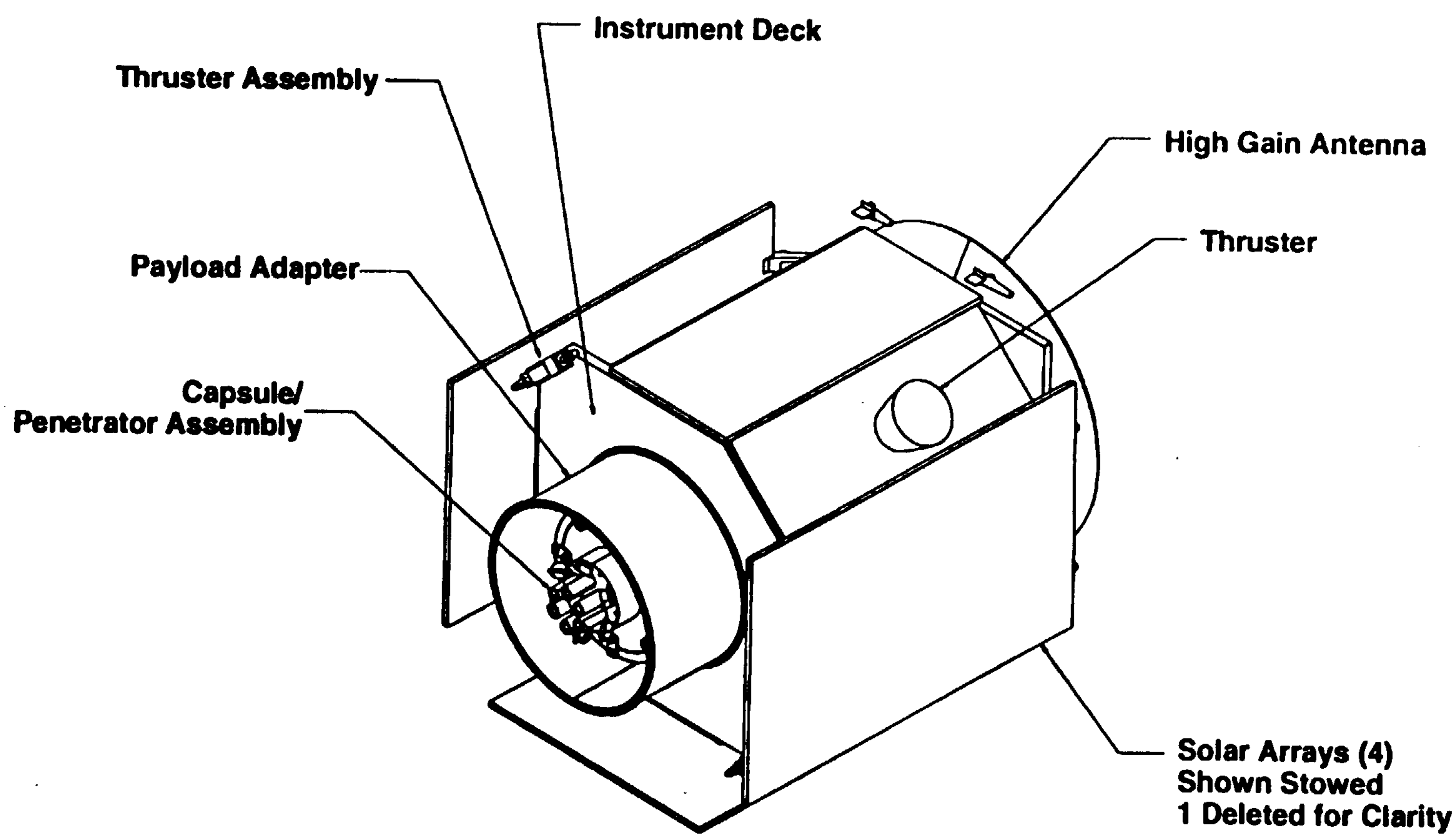


Spacecraft Launch Configuration

- High gain antenna facing upward on stack
- Return capsule on bottom of stack, mounted inside standard Delta payload attach fitting
- Solar panels stowed for launch



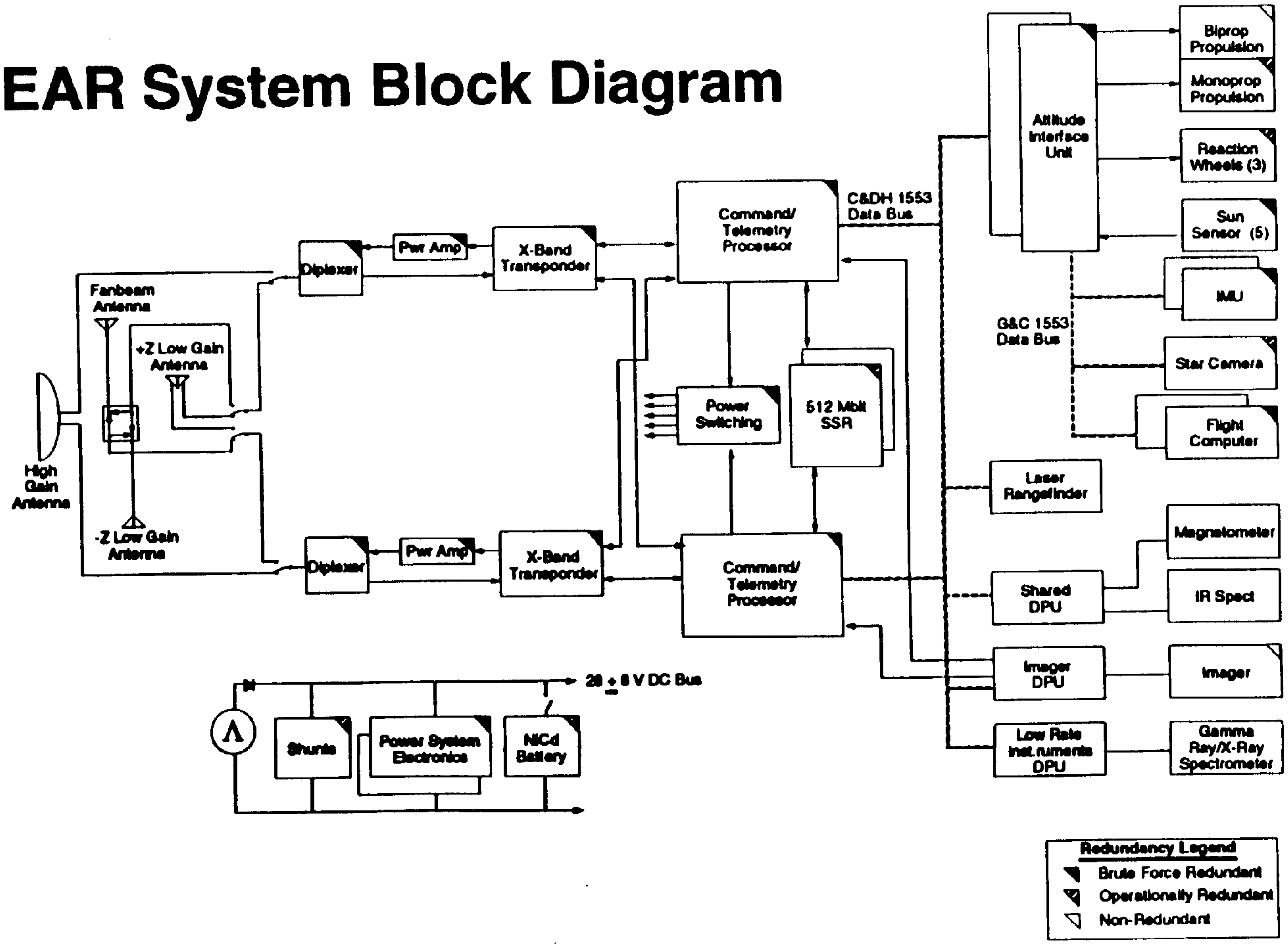
Spacecraft Launch Configuration



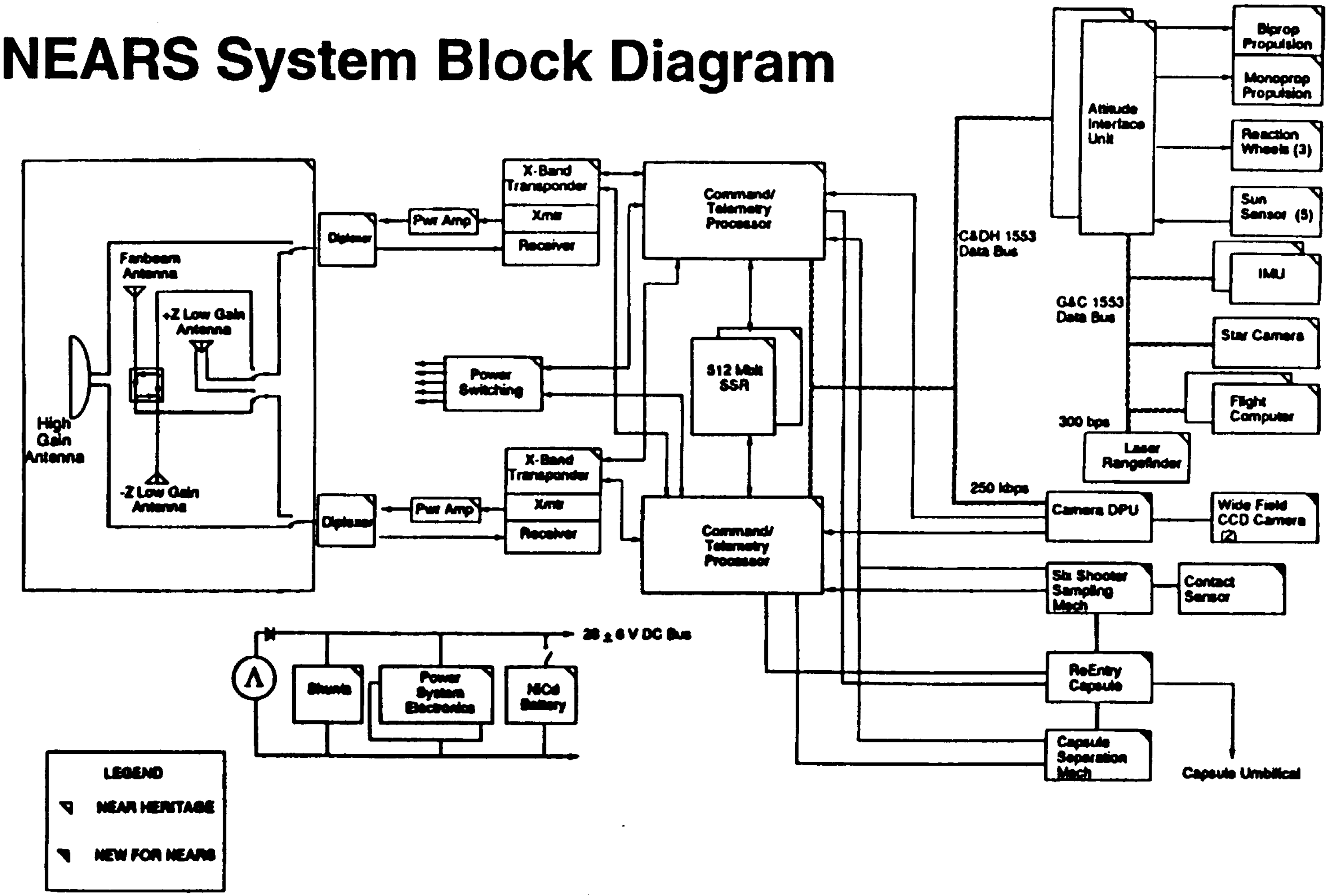
Spacecraft Design Issues

- Each subsystem must be single fault tolerant
- Single point failures minimized
- Three types of redundancy implemented:
  - **Brute Force:** Identical components to perform a single function. Examples: RF components, command system, telemetry system, reaction wheels, solid state recorders, thrusters, LGAs
  - **Internal:** Redundant systems packaged within a single unit. Examples: power switching, HGA, IMU, DSAD, solar array, motor coils
  - **Operational:** A change in operation used to perform function of failed components. Examples: power system electronics designed for 'solar only' mode if battery fails, attitude thrusters, reaction wheels compliment each other
- Launch window constraints dictate need for spares to launch on schedule
  - APL fabricated components spared at component/board level
  - Procured components spared at component level
  - Spares assembled into flight validation simulator after launch
- NEARS radiation total dose less than 15 krad
  - Most NEAR subsystems compatible without modifications.

NEAR System Block Diagram



NEARS System Block Diagram



RF Communications Subsystem						
Parabolic Antenna			5.00			NEAR
Low Gain Antenna (2)			0.60			NEAR
Fanbeam Antenna			2.00			NEAR
Power Amplifier (2)			2.52			NEAR
Transponder (2)			8.20			NEAR
DC/DC Converter (2)			3.20			NEAR
CMD Det Unit (2)			0.80			NEAR
TLM Con Unit (2)			1.80			NEAR
Diplexer (2)			0.40			NEAR
Transfer Switch			0.30			NEAR
SPDT Switches (4)			1.00			NEAR
Coax			1.25			NEAR
RF Communications Subtotal			27.07			NEAR
Attitude Determination and Control Subsystem						
Reaction Wheels (3)			7.65			NEAR
Reaction Wheel Electronics (3)			2.03			NEAR
Star Camera			2.50			NEAR
Star Camera Sunshade			0.20			NEAR
IMU (2)			5.40			NEAR
Attitude Interface Unit (2)			4.50			NEAR
DSADs (5)			1.40			NEAR
DSAD Electronic Unit			1.00			NEAR
						NEAR
A & DC Subtotal			24.68			NEAR
Command and Data Handling						
Command / Telemetry Processor (2)			12.06			NEAR
Flight Computer (2)			5.40			NEAR
Solid State Recorder (2)			3.10			NEAR
Power Switching			5.80			NEAR
C & DH Subtotal			26.36			NEAR
Thermal Subsystem						
MLI Blankets			10.00			NEAR
Heaters and Misc.			1.00			NEAR
Thermal Subtotal			11.00			NEAR
Harness Subsystem						
Harness			19.70			NEAR
Terminal Board			2.30			NEAR
Harness Subsystem			22.00			NEAR
Dry Mass (less Structure)			310.22			NEARS
Dry Mass (including Structure)			429.02			NEARS
FUEL			313.00	kg		NEARS
FUEL + 2% residual			319.26	kg		
(Max Tank Capacity = 332 kg)						
Dry Mass Contingency			43.72	kg	10.19%	
Average Specific Impulse (Secs)		299				NEAR
Delta V (m/sec)		1475				NEARS

Spacecraft Mass Summary
(NEARS Baseline)

Subsystem	Mass (kg)
Instruments	9
ReEntry Capsule	33
Sampling Mechanism	15
Propulsion (Dry)	85
Power	57
RF	27
Attitude	25
Command and Data Handling	26
Thermal	11
Harness	22
Structure	119
Dry Mass	429
Propellant	319
Total Mass	748
Launch Mass	792
Max Dry Available	473

Dry Mass Contingency **10.2%**

(Calculated as: (Launch - Total)/Max Dry)

Spacecraft Power Summary

(NEARS Baseline)

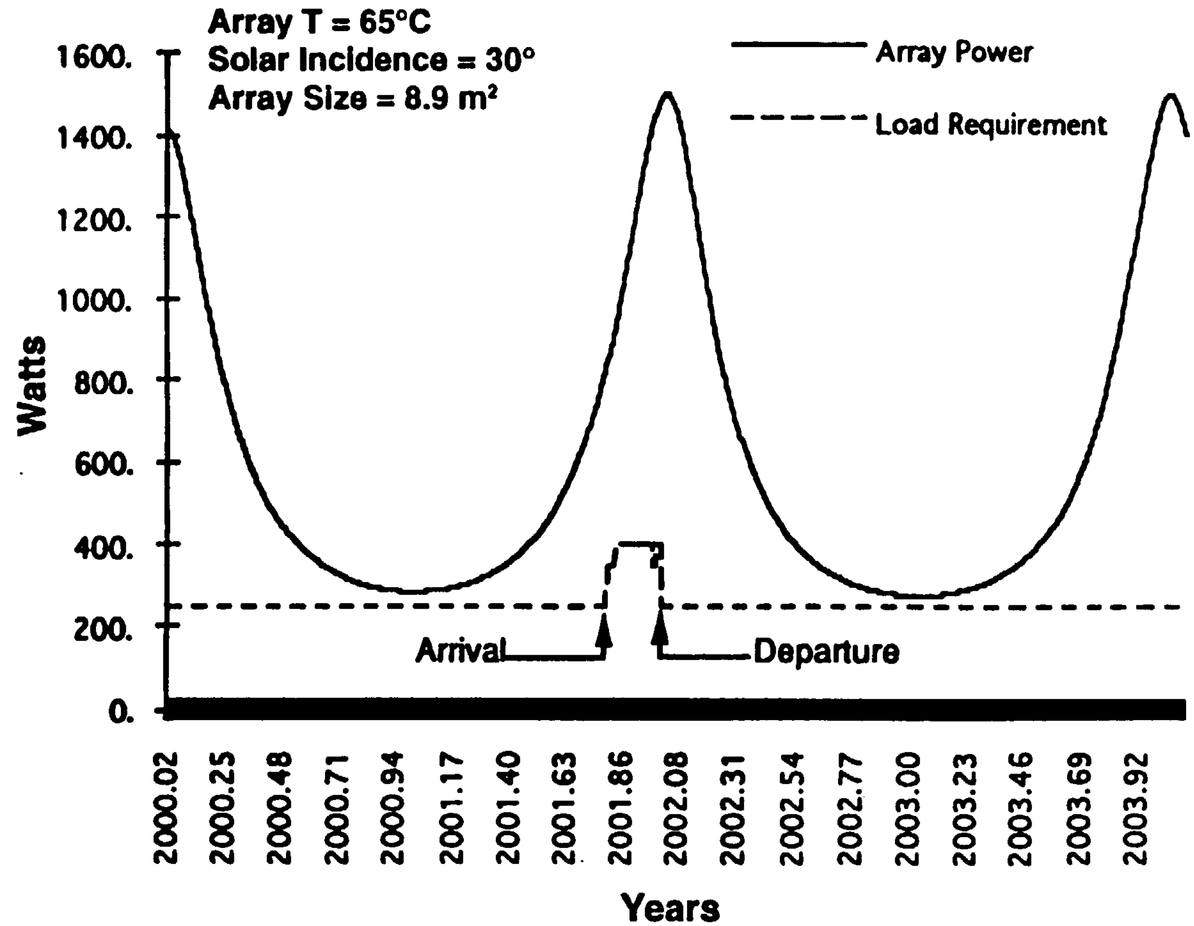
Subsystem	Cruise (Watts)	ΔV (Watts)	Asteroid (Watts)
Instruments	0.0	0.0	33.50
Propulsion	56.80	184.76	168.03
Power	13.16	13.16	13.16
RF	59.50	59.50	59.50
Attitude	61.68	35.00	84.99
Command and DH	34.85	34.85	34.85
Thermal	21.80	21.80	13.00
Total Power	248.09	349.07	407.03

Available Power @ 2.2 AU: 284 Watts

Solar Array Contingency (Cruise): 13% (Assuming a 65°C array, 30° solar incidence)

Launch Phase Battery Depth of Discharge: <20%

NEARS Baseline Power



Spacecraft Autonomy-Emergency Modes

- Autonomy performed by command system
- Two levels of low bus voltage sensing to shed loads during under voltage conditions
  - Level 1, instruments turned off, maintain attitude
  - Level 2, all non-critical loads turned off, sun pointing attitude
- Load current checking used to remove soft short circuits
- All non-critical loads fused to remove hard short circuits
- Ephemeris information periodically stored in other subsystem memory
- Each command system maintains mission clock, Sun-Earth angle data
  - Used to establish fan beam antenna use and HGA pointing in an emergency
  - Z axis must point at sun, slow rotation about sun
- Emergency command uplink possible using low gain and fan beam antenna over entire mission
  - Paired LGAs provide omni-directional coverage
  - Watchdog timer used to switch between antennas

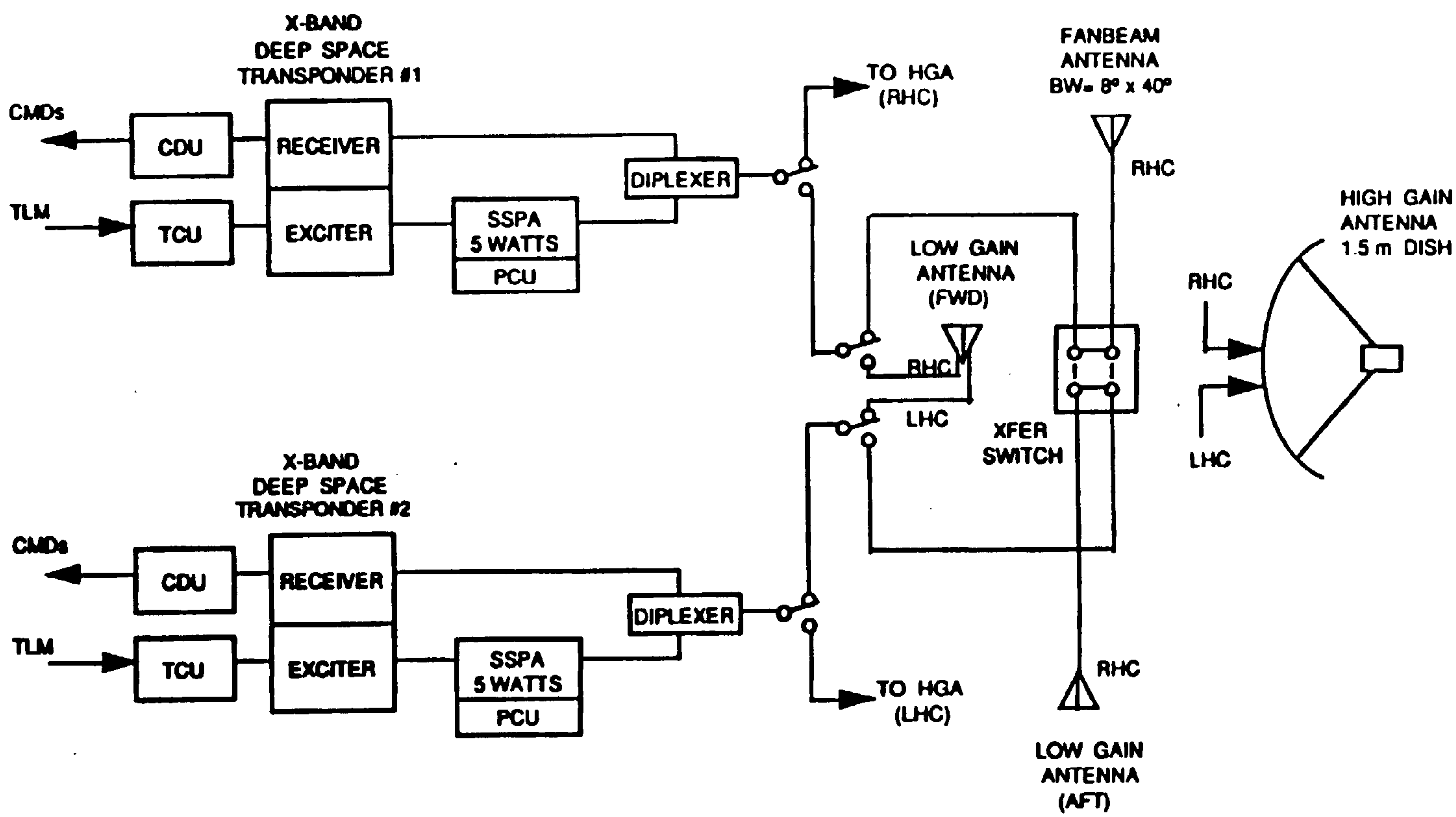
RF Communications System Requirements

- No credible single point failures
- DSN compatible
- Simultaneous X-band uplink, downlink, and tracking capabilities, except during emergency mode
- Bit-error-rates
  - Uplink Pe = 1×10^{-5}
  - Downlink Pe = 1×10^{-6}

RF Communications System Assumptions

- DSN category B mission (altitude > 2 x 10^6 km)
- Frequency bands
  - Uplink 7145-7190 MHz
  - Downlink 8400-8450 MHz
- Ground antennas
  - 34 m high efficiency antenna for all mission phases, except emergency mode downlink
  - 70 m antenna for emergency mode downlink
  - Ground antenna elevation angle = 20° minimum
- Block V receivers
- Downlink data coding: convolutional (R = 1/6, k = 15), concatenated with Reed Solomon (255,223)

RF Communication System

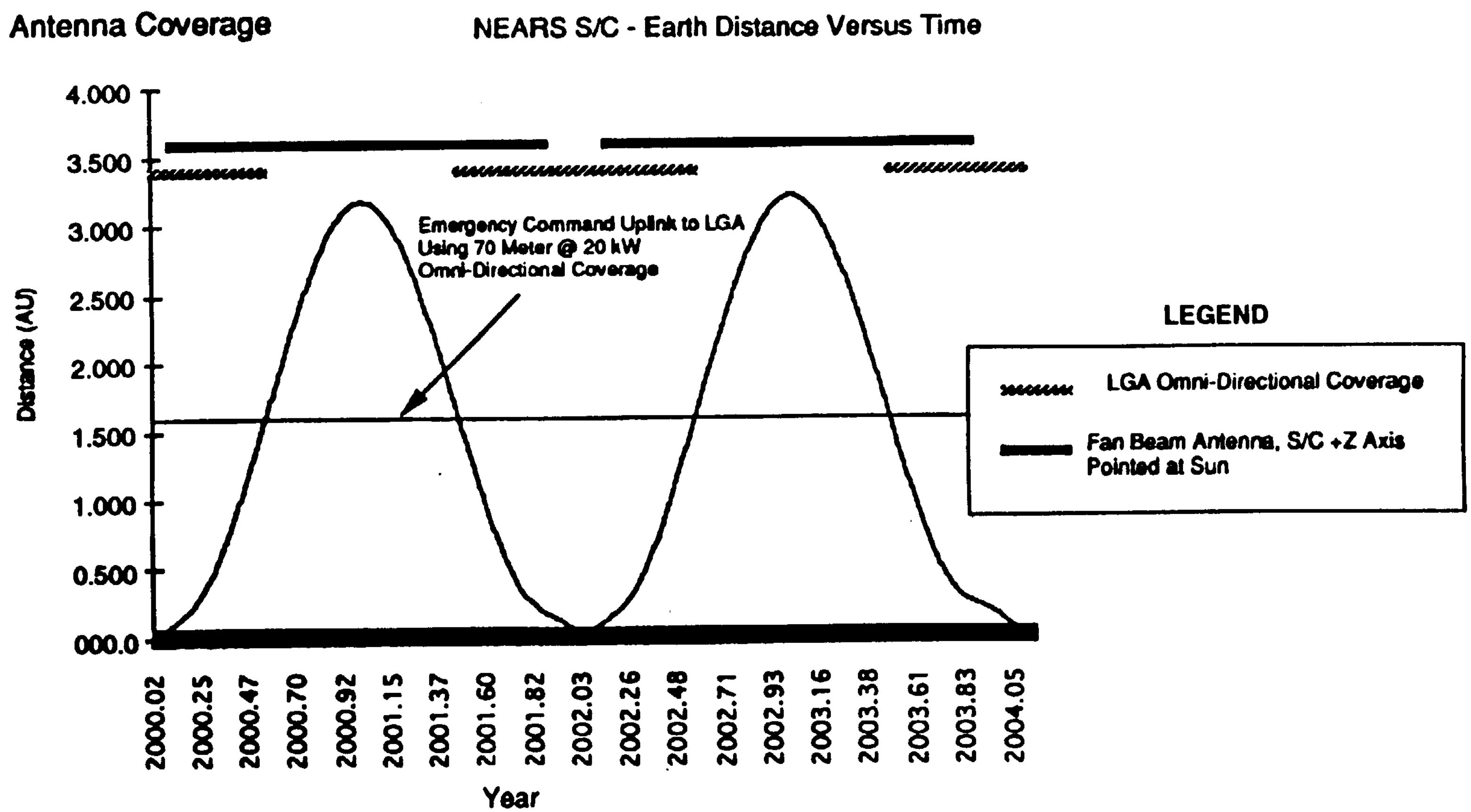


RF Communications System
Data Rates and Performance

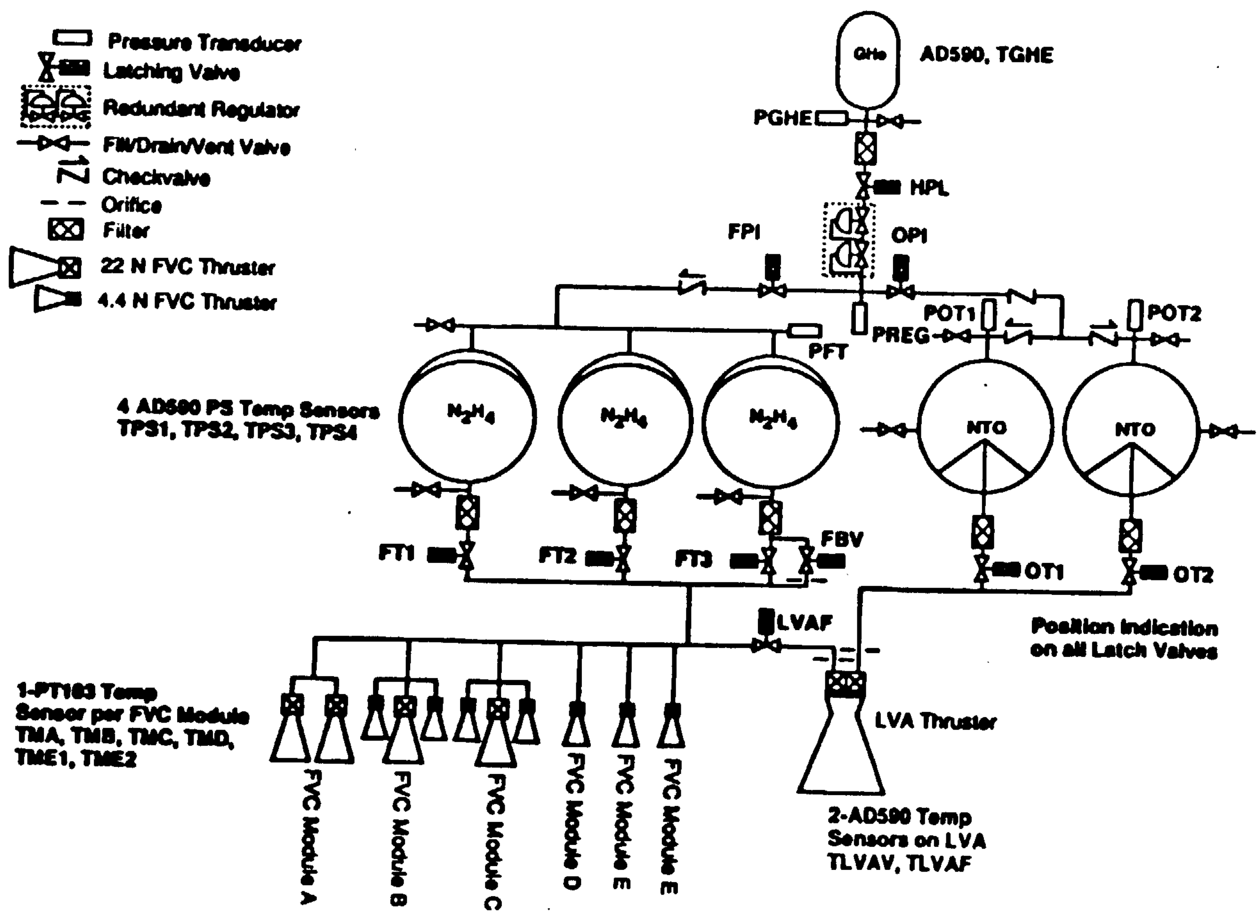
	Range (AU)	DSN Ant. (m)	S/C Ant.	Data Rate (bps)
• **Uplink**				
Emergency Mode	0-2.6	70	Fanbeam	125.0
Near Earth Mode	0-1.6	70	Broadbeam	7.83
Cruise	0-2.6	34 HEF	Hi Gain	1000
• **Downlink**				
Emergency Mode	0-2.6	70	Fanbeam	10
Near Earth Mode	0-0.16	70	Broadbeam	10
Cruise	1.2-2.6	34 HEF	Hi Gain	500
Asteroid Operations	0.1-0.3	34 HEF	Hi Gain	12-27k

Assumptions:
(1) Elevation angles ≥ 20 degrees.
(2) Ground station transmitter power = 20 kW
(3) Spacecraft transmitter = 5 Watts
(4) Data margins and carrier margins of 3 dB or greater
(5) Uplink Pe = 1×10^{-6}
(6) Downlink Pe = 1×10^{-6}
(7) 70m Goldstone Station planned upgrade to add 20 kW transmitter - complete by 5/99.

S/C - Earth Distance Versus Time



PS Hydraulic Schematic and Instrumentation



Propulsion System Design Overview

- 1 Large, bi-propthruster <314 s Isp
- 11 Small monoprop (N2H4) thrusters - ~ 234 s Isp
- 2 liquids (N2H4, N204)
  - 201 kg of hydrazine, 118 kg of oxidizer
- 1 pressurant gas (He)
- Regulated system
- CM control (within 2 cm of large thruster axis)
- Typical S/C service valves, check valves, latch valves, pyro valves and filters
- Typical S/C pressure and temperature monitoring and thermal control

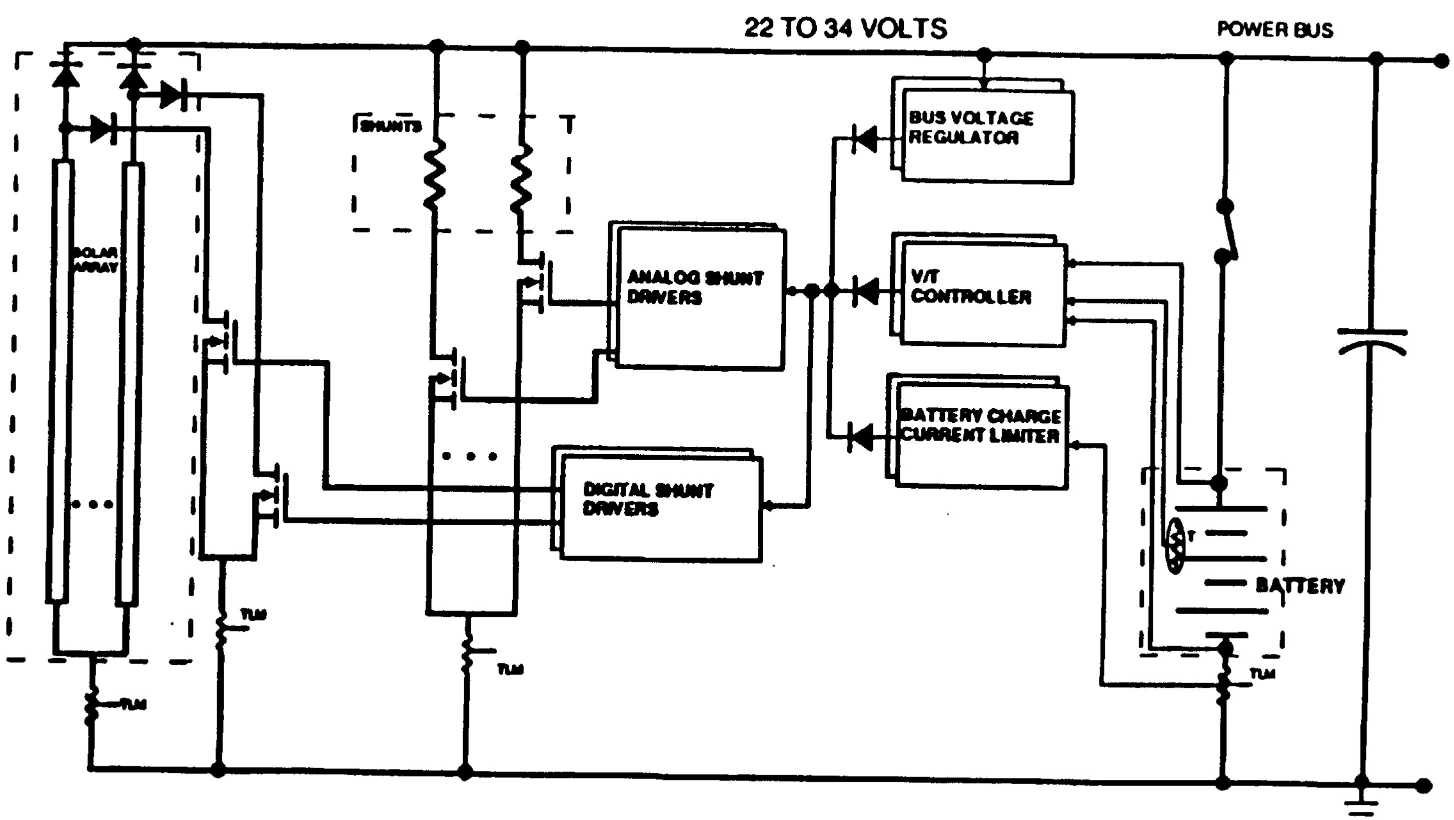
Power System Description

- Designed for 4 year mission lifetime
- Direct energy transfer (DET) topology: solar array, battery, and loads are connected directly to the bus.
- Design is based on APL NEAR power subsystem design.
- Solar array sizing based on power required for:
  - Cruise mode to 2.2 AU;
  - Asteroid rendezvous and landing operations
- Solar cells are silicon: 8 mil cells; CMX cover glass
- Shunt system used to shed excess solar array power
  - Digital shunts used for "coarse" power control
  - Analog shunts used for "fine" control
- Solar array divided into 20 segments, each with a digital shunt
  - Loss of one segment will only reduce overall power by a small amount
- Analog shunt resistors are placed outside the spacecraft to radiate heat directly to space
- Bus voltage range: With the battery on the bus: 22 to 34 volts
  "Solar only" mode, no battery: 33.5 ± 0.5 volts

Power System Description *(Cont.)*

- Battery design: 22 cell, 9 ampere-hour, advanced nickel cadmium (NiCd)
  - Used at launch, possibly at asteroid.
- Battery charge control technique used is voltage temperature (V/T) control
  - Eight ground selectable V/T curves
  - Design to support battery with one shorted cell
- Battery charge current limited to less than battery capacity C rate
- Power system electronics are redundant
- Power system fault detection and correction is performed by the C&DH system autonomy function

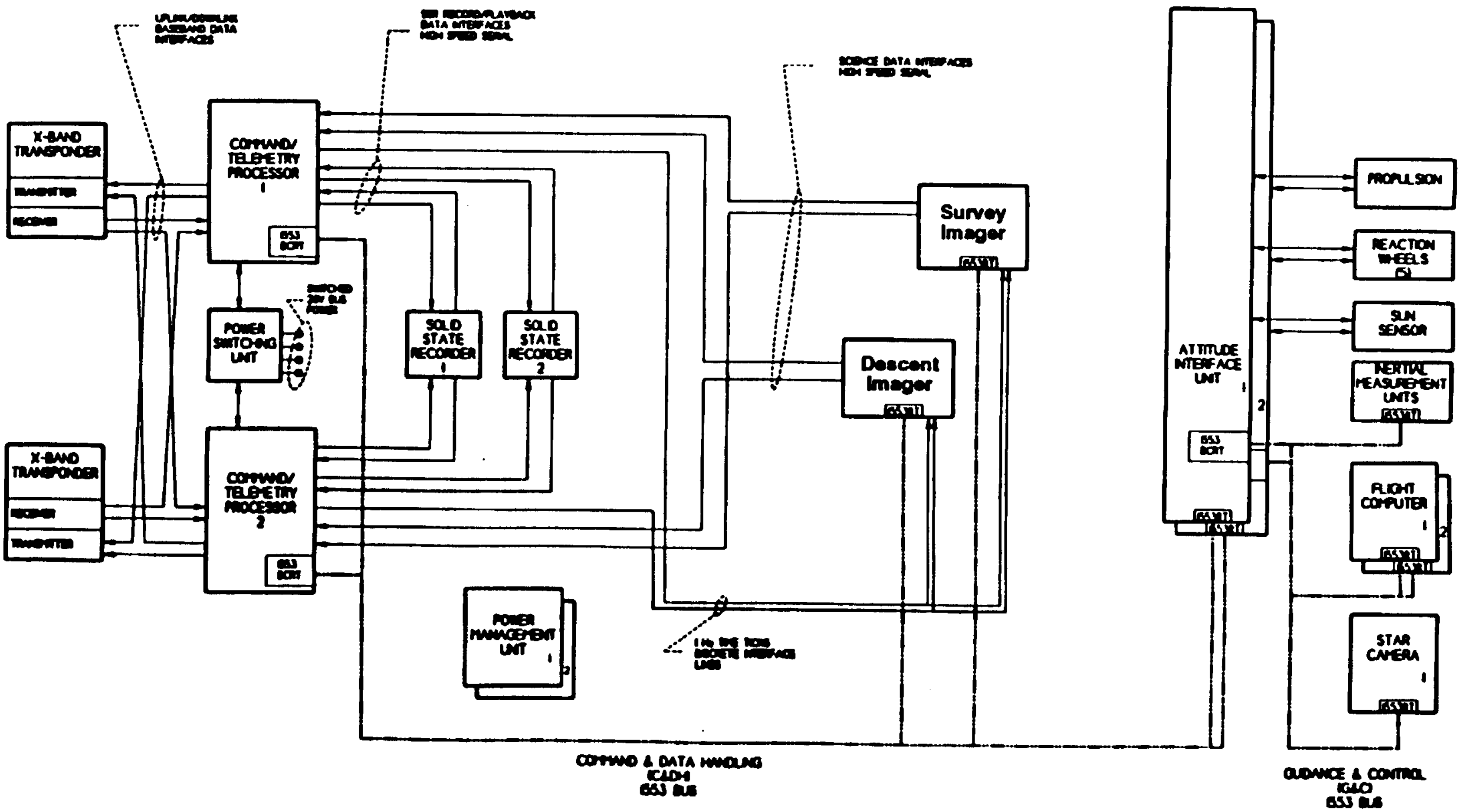
Power System Block Diagram



C&DH Functional Requirements

- Uplink command message processing
  - CCSDS, COP-1 subset
- Execution of realtime and stored commands
- Maintain and distribute spacecraft time
- Telemetry collection and processing
- Generation of downlink data streams
  - CCSDS compatible
  - Data rates to 20 kb/sec
- Onboard storage of science/engineering data
- Fault protection

Command and Data Handling Interfaces

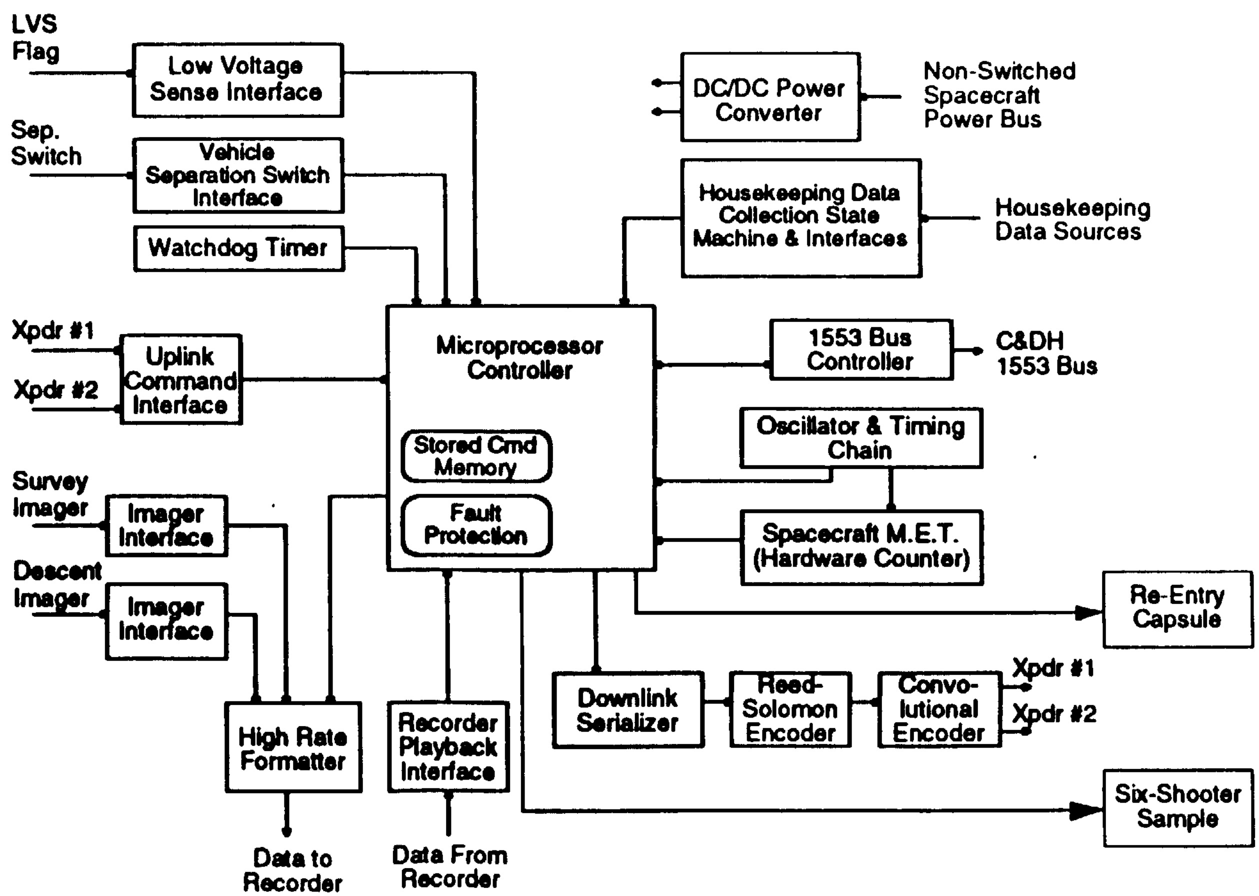


DYK-21

Command/Telemetry Processor

- Command and telemetry controlled by embedded microprocessor
- 1553 bus replaces many discrete signal lines
- Spacecraft time kept in hardware counter
  - Hardened, non-SEU susceptible logic used
  - Independent of computer software
  - Required for fault recovery
- Heritage: NEAR/ACE
- 2 flight units:
  - Power: 9.4 Watts per unit
  - Mass: 6.0 kg per unit

C/TP Diagram

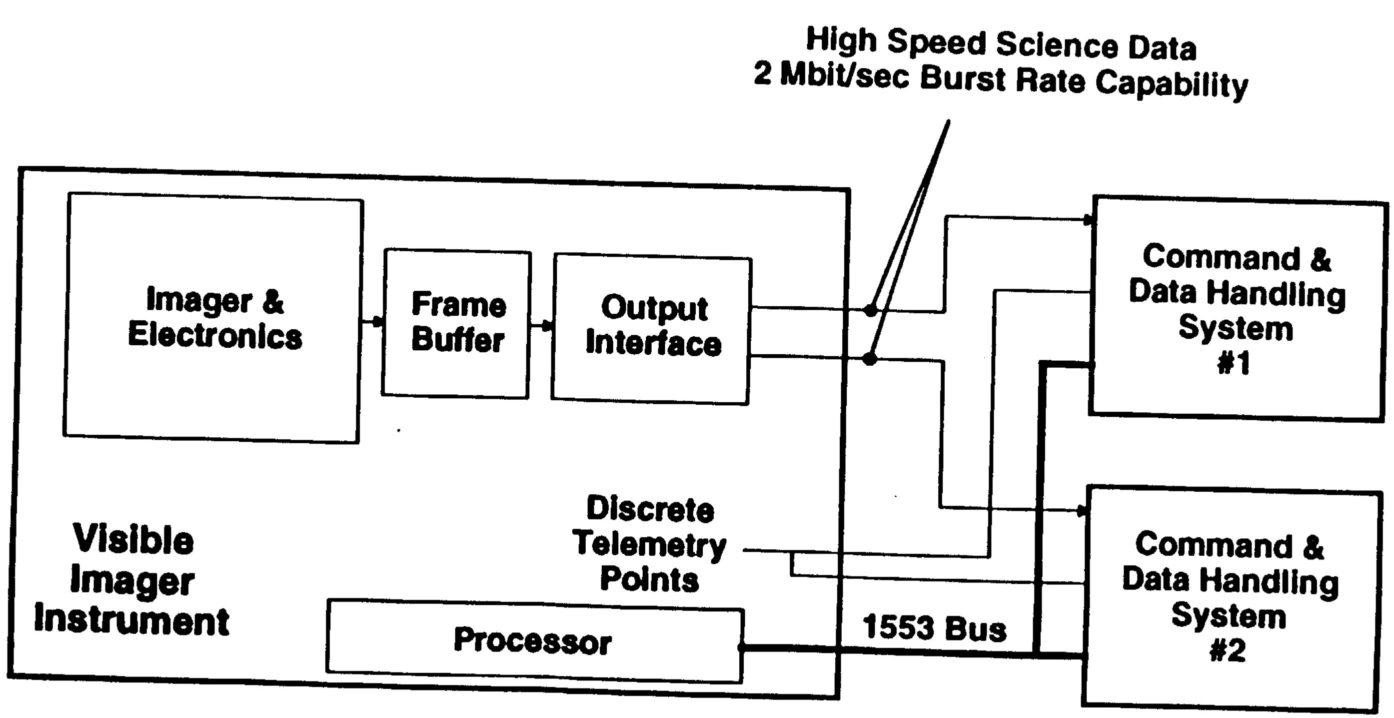


DYK-23

Science Data Interfaces

- Dedicated serial interfaces for high speed science data
  - Descent Imager
  - Survey Imager
  - 2 Mbit/sec burst rate capability
  - High speed data recorded in real time, dumped later
- 1553 bus
  - Low speed science
  - Housekeeping (engineering) data
  - Commanding

Visible Imager Interface



Solid State Recorder

- 1024 Mbit capacity (2 recorders)
- Data channels
  - 1 Record, 1 playback
  - 2 Mbit/sec data rate, clock generated by telemetry system
- Data storage organized by segments
  - Pointers available for control of record and playback
- Cross strapped I-O
  - Dual data, housekeeping, and command interfaces
- Heritage: Clementine, NEAR, ACE
- 2 flight units:
  - Power: 3.25 Watts per unit
  - Mass: 1.55 kg per unit

Power Switching Unit

- Magnetically latching relays
  - Used for controlling spacecraft bus power to switched loads
- Redundant control/status interfaces
  - Cross strapped to command telemetry processors
- Heritage: MSX, NEAR, ACE
- One Flight Unit:
  - Power: 1 Watt
  - Mass: 5.8 kg

Arc Jet Thruster Options

- Two advanced propulsion system options under study
- Potential for increased mass margin/enhanced performance
- Rocket Research Company arc jet thrusters

DYK-29T(2)

RF Communications Subsystem						
Parabolic Antenna			5.00			NEAR
Low Gain Antenna (2)			0.60			NEAR
Fanbeam Antenna			2.00			NEAR
Power Amplifier (2)			2.52			NEAR
Transponder (2)			8.20			NEAR
DC/DC Converter (2)			3.20			NEAR
CMD Det Unit (2)			0.80			NEAR
TLM Con Unit (2)			1.80			NEAR
Diplexer (2)			0.40			NEAR
Transfer Switch			0.30			NEAR
SPDT Switches (4)			1.00			NEAR
Coax			1.25			NEAR
RF Communications Subtotal			27.07			NEAR
Attitude Determination and Control Subsystem						
Reaction Wheels (3)			7.65			NEAR
Reaction Wheel Electronics (3)			2.03			NEAR
Star Camera			2.50			NEAR
Star Camera Sunshade			0.20			NEAR
IMU (2)			5.40			NEAR
Attitude Interface Unit (2)			4.50			NEAR
DSADs (5)			1.40			NEAR
DSAD Electronic Unit			1.00			NEAR
						NEAR
A & DC Subtotal			24.68			NEAR
Command and Data Handling						
Command / Telemetry Processor (2)			12.06			NEAR
Flight Computer (2)			5.40			NEAR
Solid State Recorder (2)			3.10			NEAR
Power Switching			5.80			NEAR
C & DH Subtotal			26.36			NEAR
Thermal Subsystem						
MLI Blankets			10.00			NEAR
Heaters and Misc.			1.00			NEAR
Thermal Subtotal			11.00			NEAR
Harness Subsystem						
Harness			19.70			NEAR
Terminal Board			2.30			NEAR
Harness Subsystem			22.00			NEAR
Dry Mass (less Structure)			353.80			NEARS
Dry Mass (Including Structure)			472.60			NEARS
FUEL			258.81	kg		NEARS
FUEL + 2% residual			263.98	kg		
(Max Tank Capacity = 332 kg)						
Dry Mass Contingency			55.42	kg	11.73%	
Average Specific Impulse (Secs)		380				Comsats
Delta V (m/sec)		1475				NEARS

Spacecraft Mass Summary

Option 2 (Arc Jet Thrusters)

Subsystem	Mass (kg)
Instruments	9
ReEntry Capsule	33
Sampling Mechanism	15
Propulsion (Dry)	72
Power	114
RF	27
Attitude	25
Command and Data Handling	26
Thermal	11
Harness	22
Structure	119
Dry Mass	473
Propellant	264
Total Mass	737
Launch Mass	792
Max Dry Available	528

Dry Mass Contingency 11.6%

(Calculated as: (Launch - Total)/Max Dry)

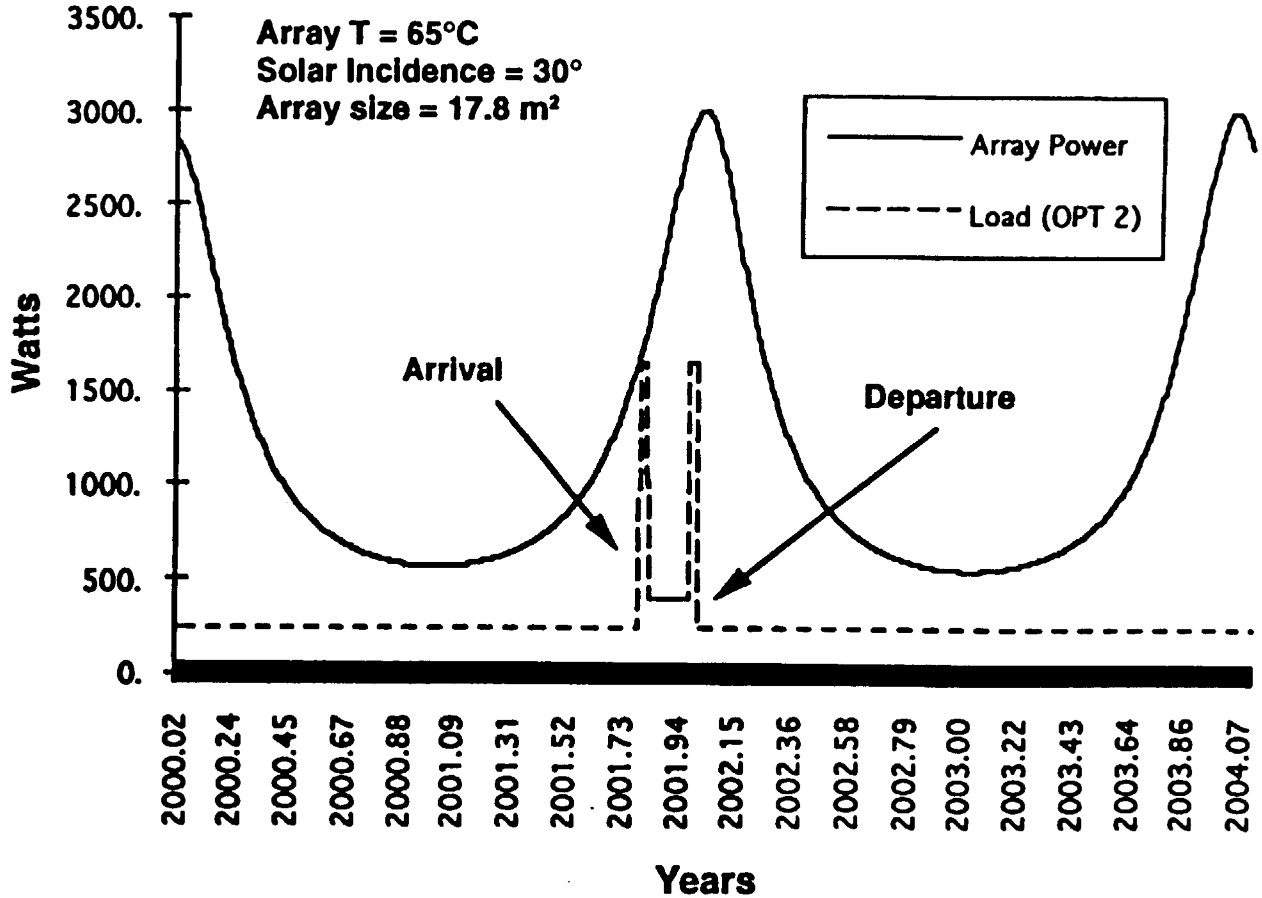
C-2

Spacecraft Power Summary
Option 2 (Arc Jet Thrusters)

Subsystem	Cruise (Watts)	ΔV (Watts)	Asteroid (Watts)
Instruments	0.0	0.0	33.50
Propulsion	56.80	1484.76	168.03
Power	13.16	13.16	13.16
RF	59.50	59.50	59.50
Attitude	61.98	35.00	84.99
Command and DH	34.85	34.85	34.85
Thermal	21.80	21.80	13.00
Total Power	248.09	1649.00	407.03

NEARS Power Profile

Option 2 (Arc Jet Thrusters)



DYK-32T(2)

RF Communications Subsystem						
Parabolic Antenna			5.00			NEAR
Low Gain Antenna (2)			0.60			NEAR
Fanbeam Antenna			2.00			NEAR
Power Amplifier (2)			2.52			NEAR
Transponder (2)			8.20			NEAR
DC/DC Converter (2)			3.20			NEAR
CMD Det Unit (2)			0.80			NEAR
TLM Con Unit (2)			1.80			NEAR
Diplexer (2)			0.40			NEAR
Transfer Switch			0.30			NEAR
SPDT Switches (4)			1.00			NEAR
Coax			1.25			NEAR
RF Communications Subtotal			27.07			NEAR
Attitude Determination and Control Subsystem						
Reaction Wheels (3)			7.85			NEAR
Reaction Wheel Electronics (3)			2.03			NEAR
Star Camera			2.50			NEAR
Star Camera Sunshade			0.20			NEAR
IMU (2)			5.40			NEAR
Attitude Interface Unit (2)			4.50			NEAR
DSADs (5)			1.40			NEAR
DSAD Electronic Unit			1.00			NEAR
						NEAR
A & DC Subtotal			24.68			NEAR
Command and Data Handling						
Command / Telemetry Processor (2)			12.06			NEAR
Flight Computer (2)			5.40			NEAR
Solid State Recorder (2)			3.10			NEAR
Power Switching			5.80			NEAR
C & DH Subtotal			26.36			NEAR
Thermal Subsystem						
MLI Blankets			10.00			NEAR
Heaters and Misc.			1.00			NEAR
Thermal Subtotal			11.00			NEAR
Harness Subsystem						
Harness			19.70			NEAR
Terminal Board			2.30			NEAR
Harness Subsystem			22.00			NEAR
Dry Mass (less Structure)			316.66			NEARS
Dry Mass (Including Structure)			435.46			NEARS
FUEL			234.16	kg		NEARS
FUEL + 2% residual			238.84	kg		
(Max Tank Capacity = 332 kg)						
Dry Mass Contingency			117.70	kg	27.03%	
Average Specific Impulse (Secs)		429				NEARS
Delta V (m/sec)		1475				NEARS

Spacecraft Mass Summary

Option 3 (Advanced Arc Jet Thrusters)

Subsystem	Mass (kg)
Instruments	9
ReEntry Capsule	33
Sampling Mechanism	15
Propulsion (Dry)	72
Power	77
RF	27
Attitude	25
Command and Data Handling	26
Thermal	11
Harness	22
Structure	119
Dry Mass	436
Propellant	239
Total Mass	675
Launch Mass	792
Max Dry Available	553

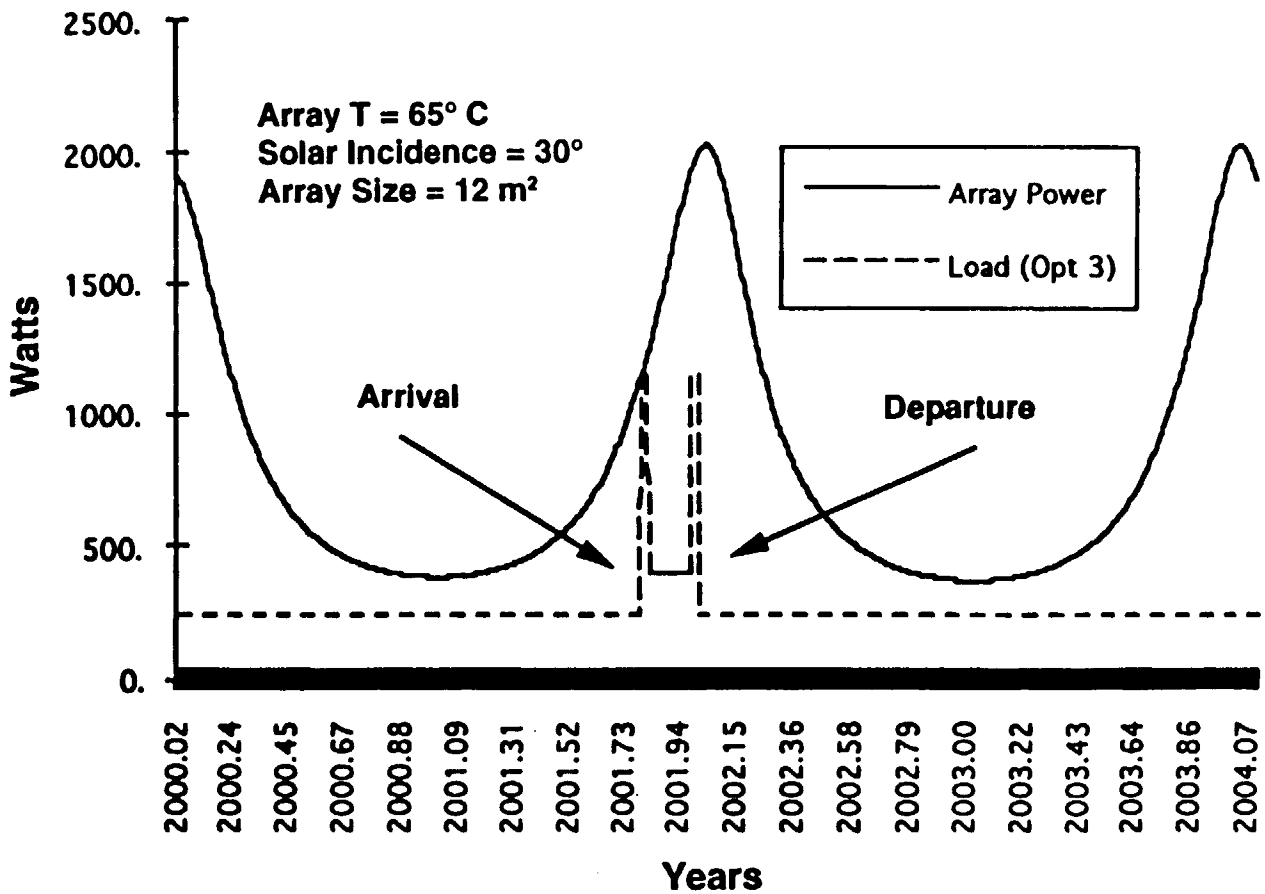
Dry Mass Contingency 26.8%
(Calculated as: (Launch - Total)/Max Dry)

Spacecraft Power Summary
Option 3 (Advanced Arc Jet Thrusters)

Subsystem	Cruise (Watts)	ΔV (Watts)	Asteroid (Watts)
Instruments	0.0	0.0	33.50
Propulsion	56.80	984.76	168.03
Power	13.16	13.16	13.16
RF	59.50	59.50	59.50
Attitude	61.98	35.00	84.99
Command and DH	34.85	34.85	34.85
Thermal	21.80	21.80	13.00
Total Power	248.09	1149.07	407.03

NEARS Power Profile

Option 3 (Advanced Arc Jet Thrusters)



Touch-and-Go-Sampling

- No long duration landing
- Spacecraft makes momentary contact with asteroid and obtains sample with pyrotechnic device that fires sampling tube into surface
- Sample can be obtained in zero gravity
- Sample can be obtained from rock or regolith surface
- Pyrotechnic sampling device is under development at APL and has operated successfully into concrete and sand targets
- Accommodate a cluster of sampling tubes ("six-shooter") for multiple samples

Touch and Go Sampling

Quasi-Vertical Descent

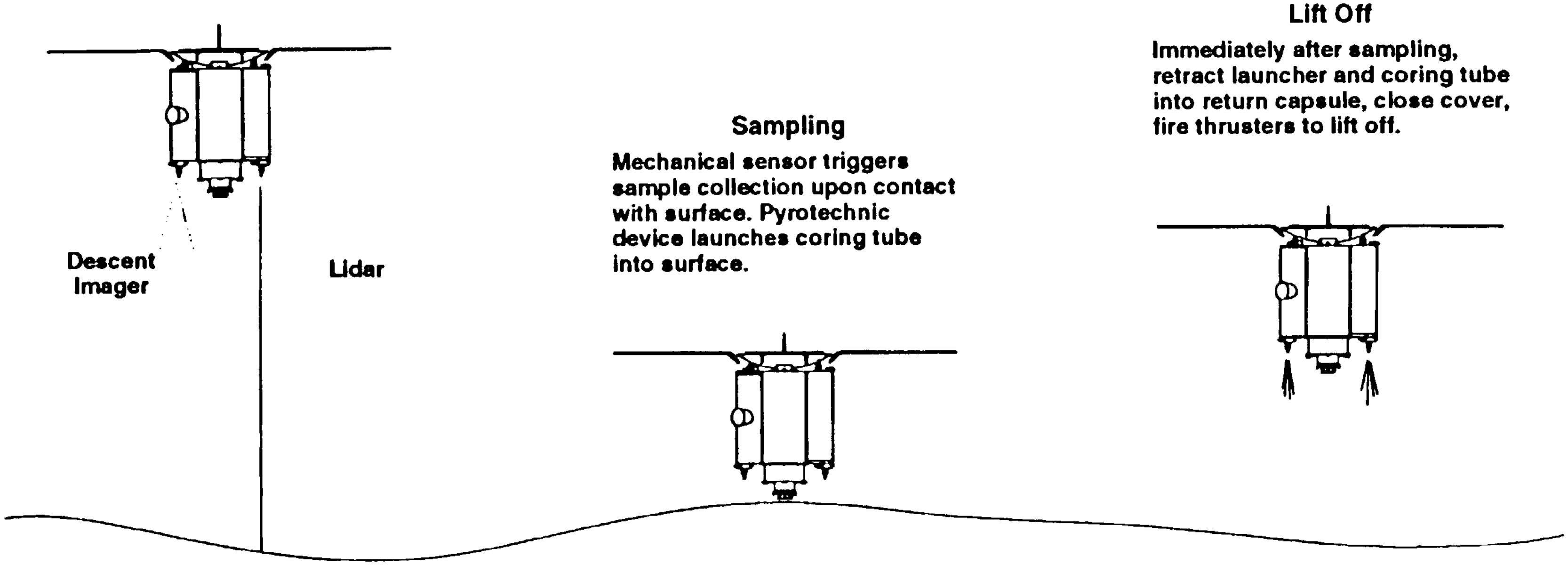
From rendezvous orbit, fire thrusters to enter autonomous, nearly vertical descent as seen from asteroid. Terminal braking burns under altimeter control.

Sampling

Mechanical sensor triggers sample collection upon contact with surface. Pyrotechnic device launches coring tube into surface.

Lift Off

Immediately after sampling, retract launcher and coring tube into return capsule, close cover, fire thrusters to lift off.



Six-Shooter Sample Collector

- NEARS six-shooter will accommodate six individual launchers and core tubes
- NEARS core tube and launcher systems will be tested into rock, regolith, and meteorite targets to verify capability to acquire > 10 gm samples and retention of core tube by launcher when fired into soft targets
- There is no requirement to retract core tube into launcher after firing
- Entire launcher, with core tube extended and sample inside, retracts into return capsule after firing

Six-Shooter Sample Collector *(Cont.)*

Technical Requirements:

- Small size and mass
- Minimum power to be supplied by carrier vehicle
- Minimum sustained force from carrier vehicle
- Operation in near vacuum and microgravity
- Safe and reliable operation

Scientific Requirements:

- Collection of 10 to 100 grams of sample mass, from solid rock or regolith
- Minimal alteration and contamination of sample

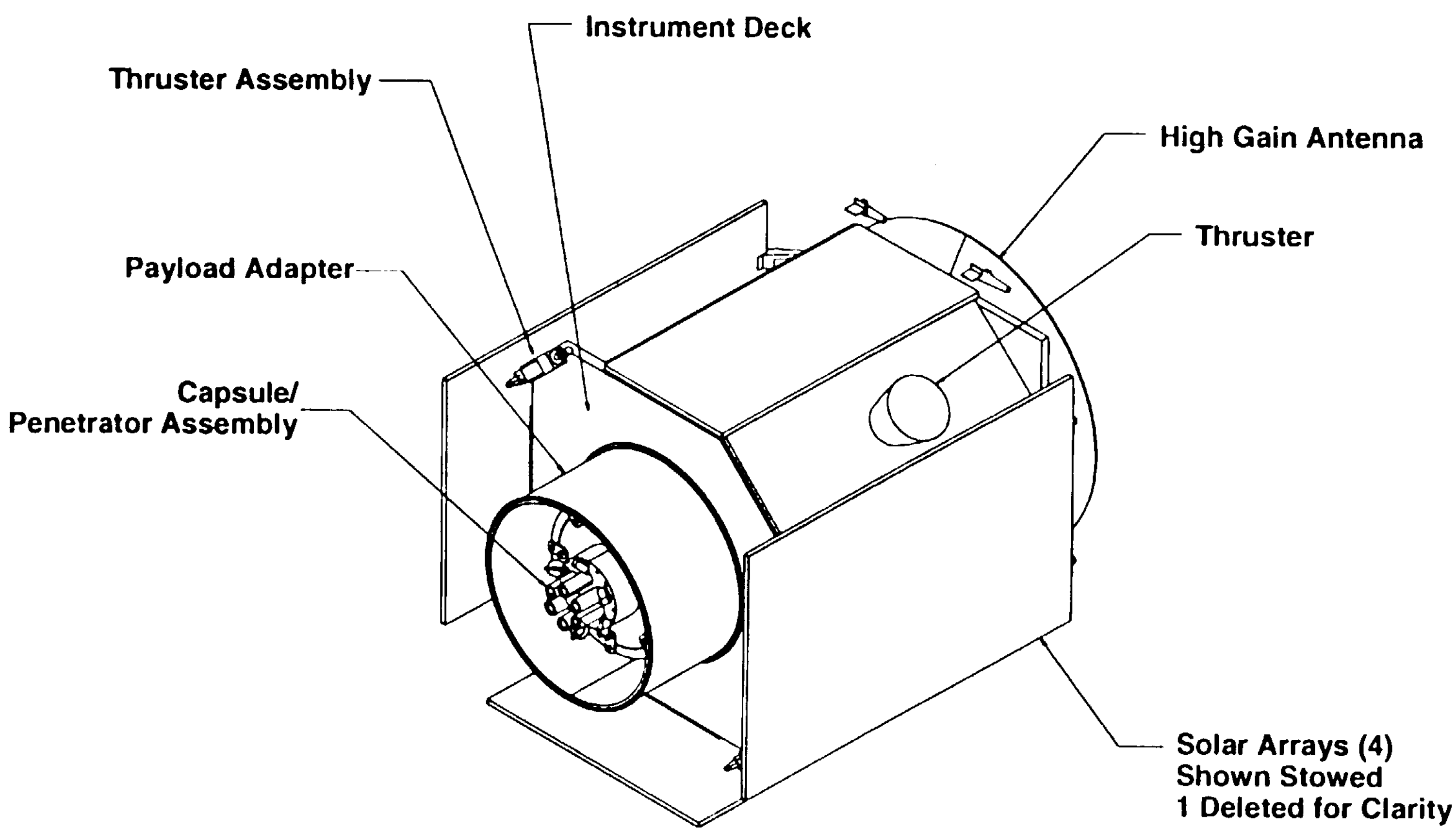
Six-Shooter Sample Collector *(Cont.)*

- Pyrotechnic gas generator propels a coring tube into surface
- Launcher maintains alignment of corer and has mechanical stop to limit its travel if surface does not do so
- Propellant gases maintained completely separate from sample
- Corer creates a crater and does not become bound to surface
- Concept of operation is similar to that of side-wall samplers used in terrestrial drilling operations

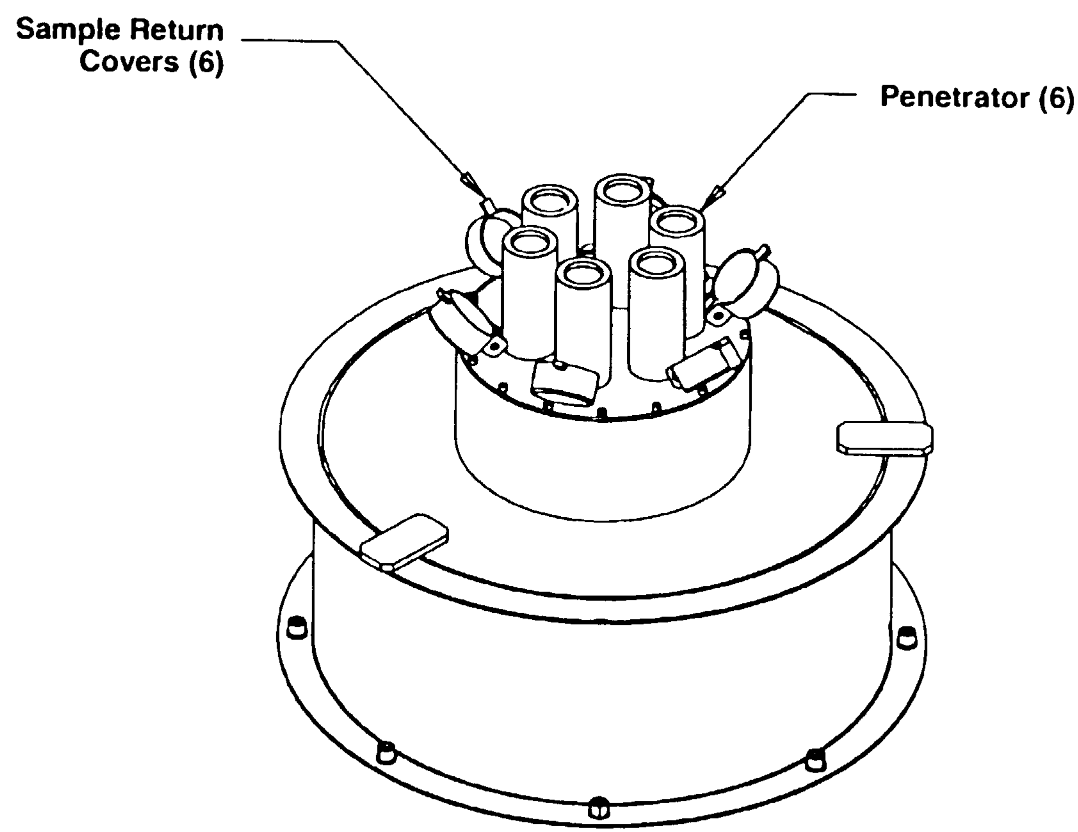
Sampling Handling Onboard Spacecraft

- Earth return capsule is mounted with aft end facing away from spacecraft
- Six-shooter sample collector is mounted within the return capsule and operates from within it
- After each core tube is fired, its launcher (together with core tube and sample) retracts into the return capsule and its individual cover closes
- Entire sample collector, with each sample individually sealed, is returned to Earth
- There is no robotic manipulator arm for sample handling

Spacecraft Launch Configuration

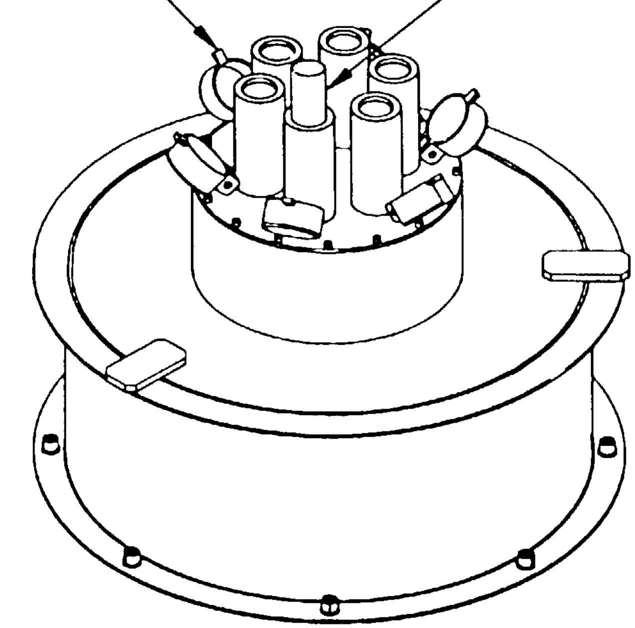


Six-Shooter and Return Capsule Configuration



Six-shooter ready to fire

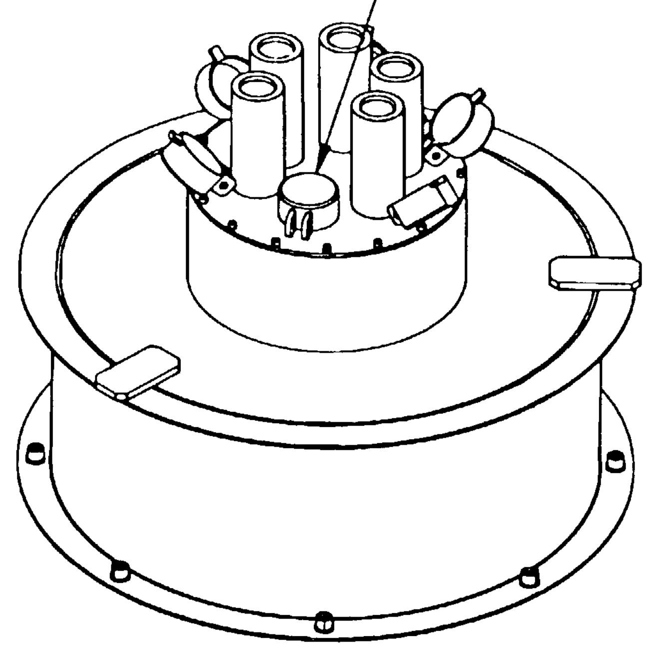
Six-Shooter and Return Capsule Configuration *(Cont.)*



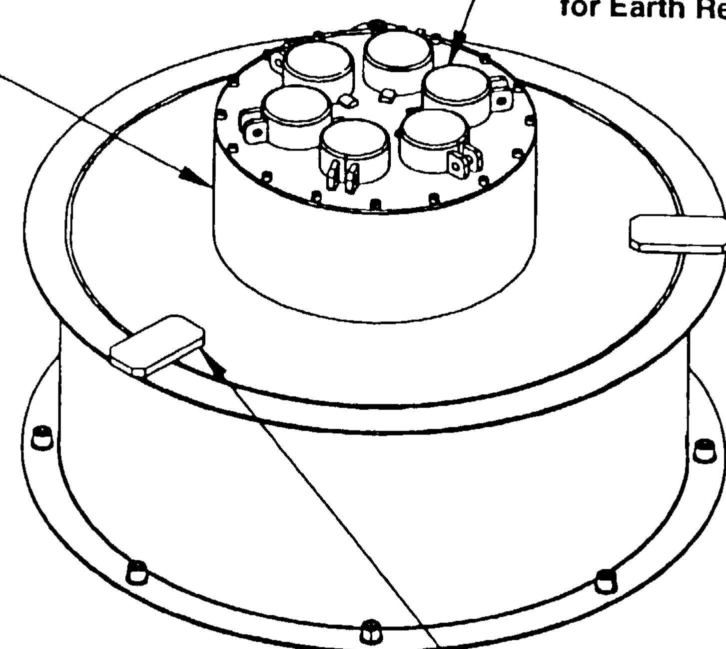
One sampler fired

Six-Shooter and Return Capsule Configuration *(Cont.)*

After firing, entire launcher (with penetrator extended) retracts into the return capsule and a cover closes.

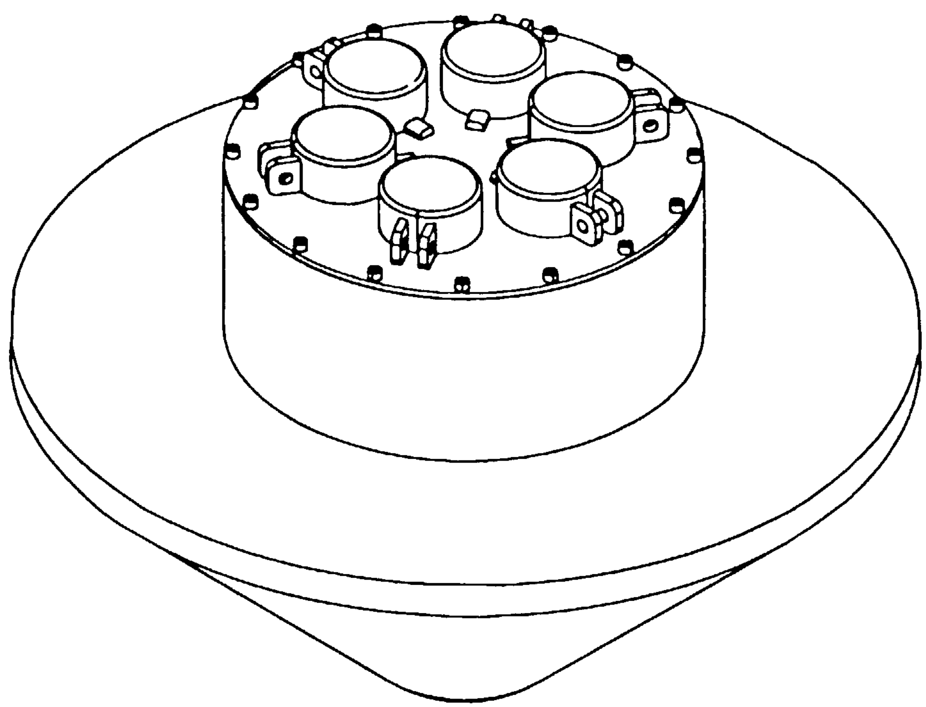


Six-Shooter and Return Capsule Configuration *(Cont.)*



All lauchers have been fired

Six-Shooter and Return Capsule Configuration *(Cont.)*



Configuration after separation

Landing Scenario

- Establish retrograde parking orbit at 10-15 body radii
- Enter elliptical transfer orbit with periapse at 2 body radii
- Near periapse of transfer orbit, enter quasi-vertical descent
- Quasi-vertical descent trajectory determined from asteroid rotation state, shape and gravity field under ground control
- Terminal braking burn under altimeter control
- Velocity components relative to asteroid surface at contact can be <10 cm/s vertical, <1 cm/s horizontal
- Mechanical contact sensor triggers firing of sample collector and initiates liftoff burn

Landing Scenario *(Cont.)*

- Solar powered spacecraft, with fixed instruments, fixed antennas, fixed solar panels, and fixed sample collectors, cannot land anywhere on asteroid at any time
- Landing must occur on dayside, unless battery is used
- Accessible portion of asteroid surface depends on rotation pole which will be determined early during rendezvous phase
- Spacecraft maintains fixed inertial attitude during quasi-vertical descent, with aft end (-z) toward surface, fanbeam antenna to Earth, solar panels within 70° of full illumination
- Spacecraft maintains continuous telemetry link to Earth during descent and touchdown

The Problem

- **Objective:**
  - Soft landing on asteroid with minimal complexity and cost
- **Heritage:**
  - Surveyor - define "soft" as 0.1-1.0 m/sec vertical velocity with energy absorbing pads, and some bounce allowed
  - NEAR - stability of retrograde orbits
- **Inputs:**
  - NEAR analysis shows orbit can be navigated with accuracy of several meters in position, several mm/sec in velocity.
  - Open loop control is cheap. Closed loop control using an altitude only sensor is also relatively inexpensive.

Analysis Tools

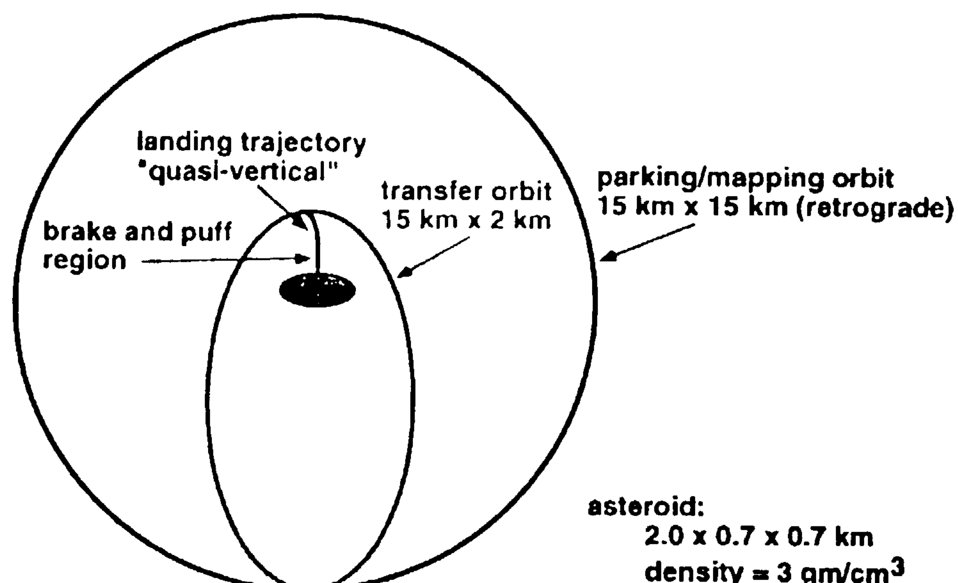
- Asteroid Landing
  - Asteroid landing simulator
  - Written in Manugistic APL
  - Uses finite mascon model of gravity field
  - Uses ellipsoidal surface model
  - Performs 4th order Runge-Kutta integration
  - Allows for open-loop thruster program
  - Can be expanded into closed-loop control
- Landing Solver
  - Determines thrust program necessary to transition between approach and landing trajectories
  - Written in Manugistic APL

Stressing Case

- Landing site choice will be influenced by several factors:
  - Nature of satellite surface
  - Visibility of landing site from earth
  - Solar illumination of landing site
  - Visibility of landing trajectory from earth
- These factors may necessitate landing at various "latitudes."
- The most stressing case (from a landing viewpoint) is landing at a low point on the equator.
- Therefore, we have begun our study with this case.

Approach and Landing Concept

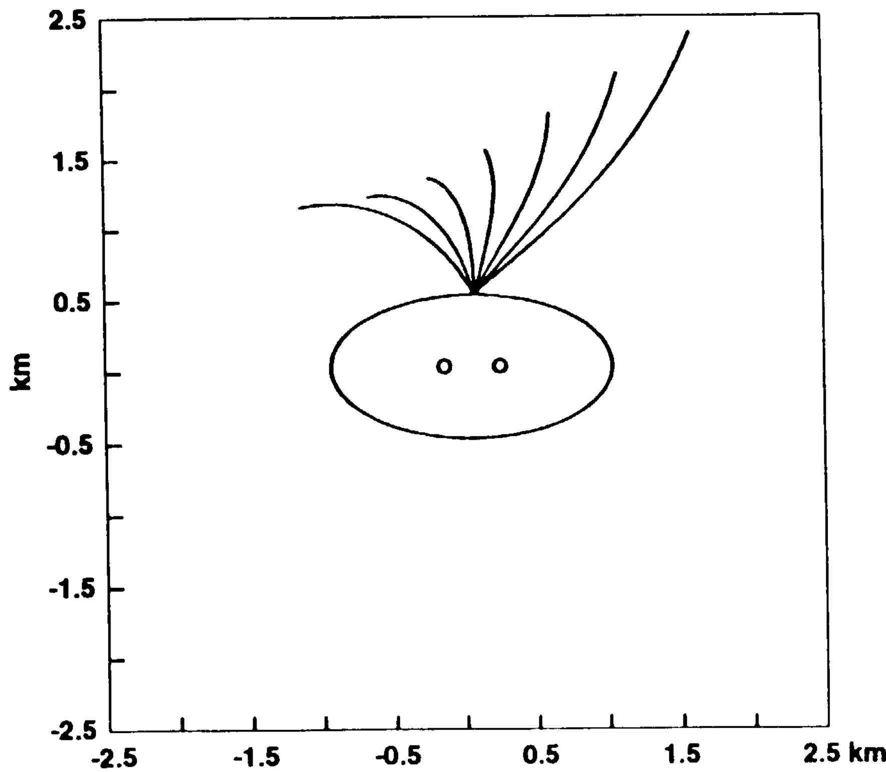
- Begin in circular parking/mapping orbit
- Descend through open loop command of two thruster firings



asteroid:
2.0 x 0.7 x 0.7 km
density = 3 gm/cm^3
10 hour period

Quasi-Vertical Trajectory

- It is possible to choose a landing trajectory that is nearly vertical in the body-fixed frame of the asteroid, although this trajectory will have non-zero horizontal velocity when vertical-only braking is applied.



Asteroid:
- shape: 2.0 x 1.0 x 1.0 km
- rotation rate: 10 hours

Landing:
- vertical velocity: 1.23 m/sec
- horizontal velocity:
  - nominal: 0.33 m/sec
  - range: -0.66 to 1.32 m/sec

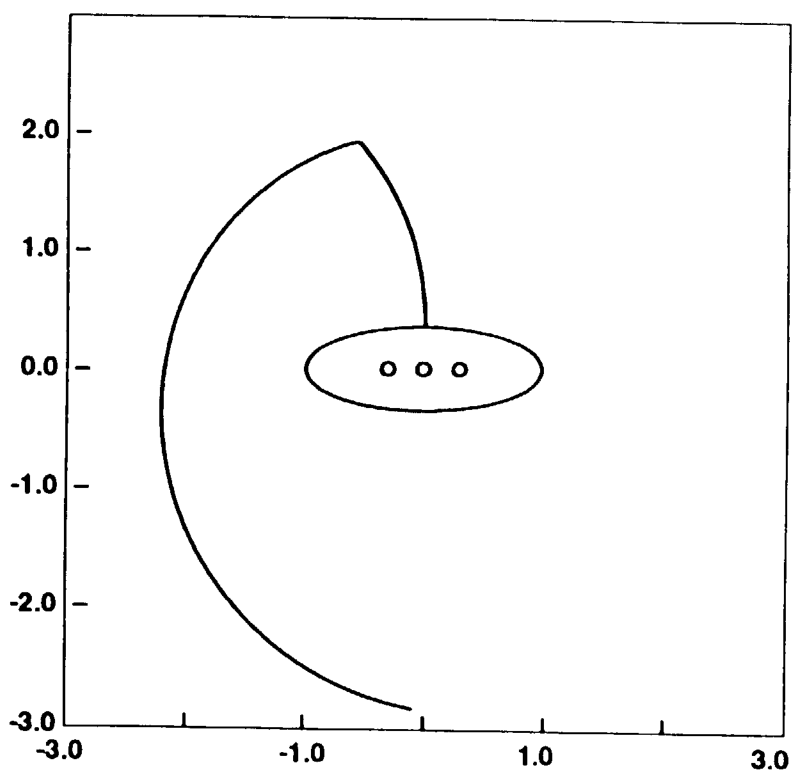
Baseline Landing Concept (Closed Loop)

- Choose quasi-vertical velocity trajectory with zero parallel velocity at touchdown:
  - It does not require non-vertical thrusting (which simplifies landing control problem)
- Altimeter measures range and range rate to surface
- Use sequence of braking burns triggered by altimeter
- Landing at 0.1 m/sec vertical velocity, 0 m/sec horizontal velocity with closed loop control

Backup Landing Concept (Open Loop)

- Choose quasi-vertical trajectory with zero parallel velocity at touchdown
- Use predetermined single vertical braking burn followed by a sequence of evenly spaced decelerating "puffs"
- Landing at <1 m/sec vertical velocity with open loop control

Typical Landing Sequence



Asteroid:
- diameters: 2.0 x 0.7 x 0.7 km
- rotation rate: 10 hours
- mass = 1.5×10^{12} kg
- 3 mascons:
  - 50% central
  - 25% at ± 0.3 km

Transfer orbit:
- 2 x 15 km

Transfer to descent trajectory:
- single 0.81 m/sec burn

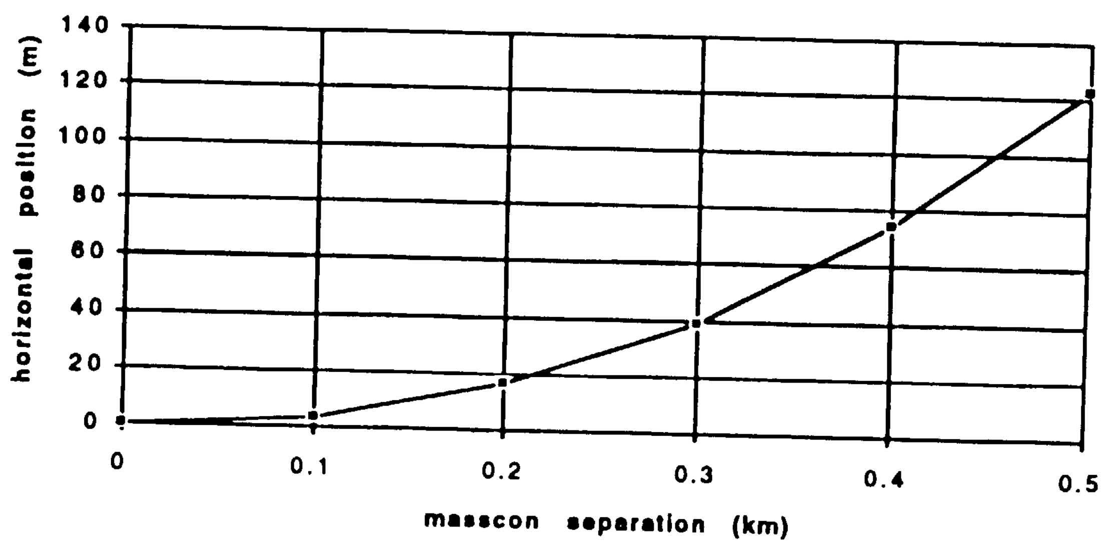
Descent:
- braking burns under closed loop control, or single 1 m/sec braking burn followed by continual 0.015 m/sec "puffs" at 60 sec intervals (open loop)

Sources of Error

- What are the principal sources of error?
- Which error sources dominate?
- Will errors be small enough to allow open loop control?

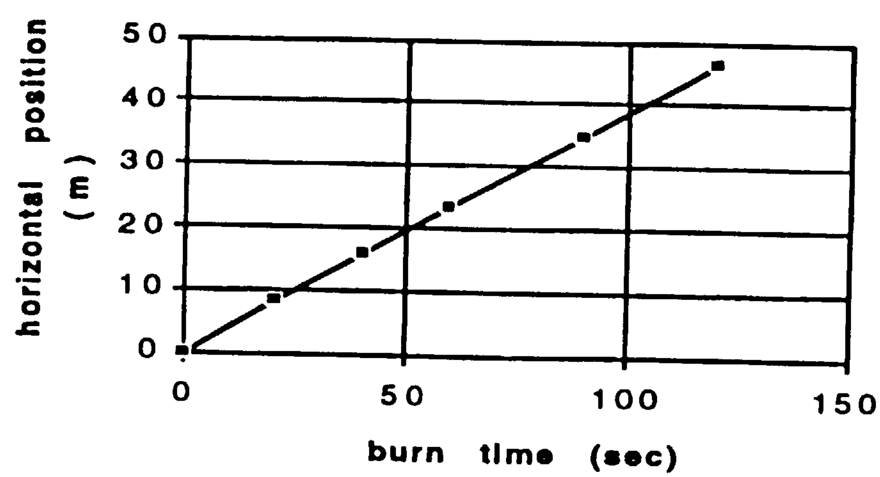
Effect of Gravity Model Error on Landing Position

- Nominal case:
  - Asteroid mass = 10^{13} kg
  - Asteroid dimensions = 1.0 x 0.5 x 0.5 km
  - Rotational period = 10 hours
  - Landing approach = single impulsive burn from 1.5 x 30 km transfer orbit



Effect of Burn Time on Landing Position

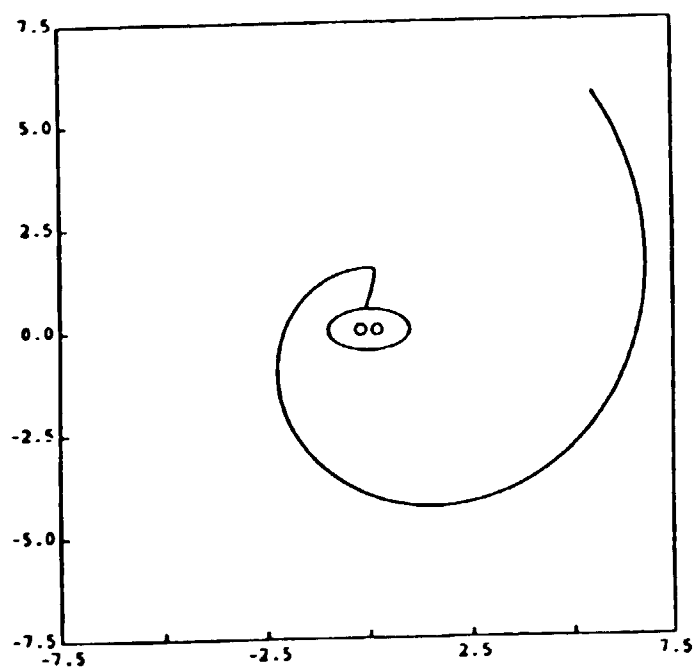
- **Nominal case:**
  - Asteroid mass = 10^{13} kg
  - Asteroid dimensions = 1.0 x 0.5 x 0.5 km
  - 2 mascons, 0.4 km apart
  - Rotational period = 10 hours
  - Landing approach = single non-impulsive burn from 1.5 x 30 km transfer orbit

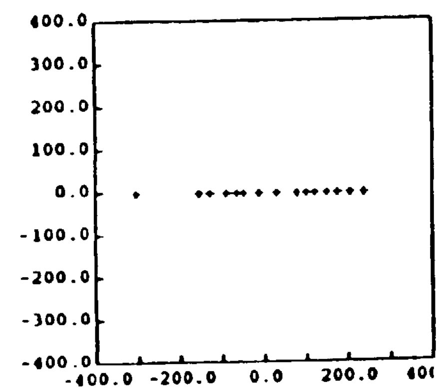


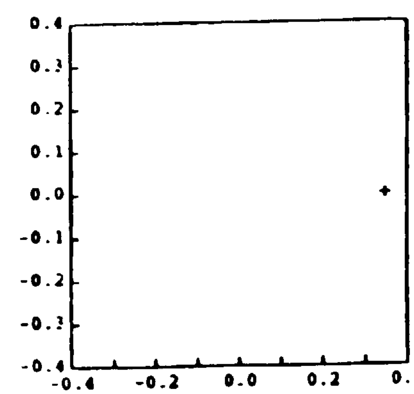
Effect of Burn ΔV Error on Landing Position

Error = 0.1 m/sec (1 s)

ID: 130

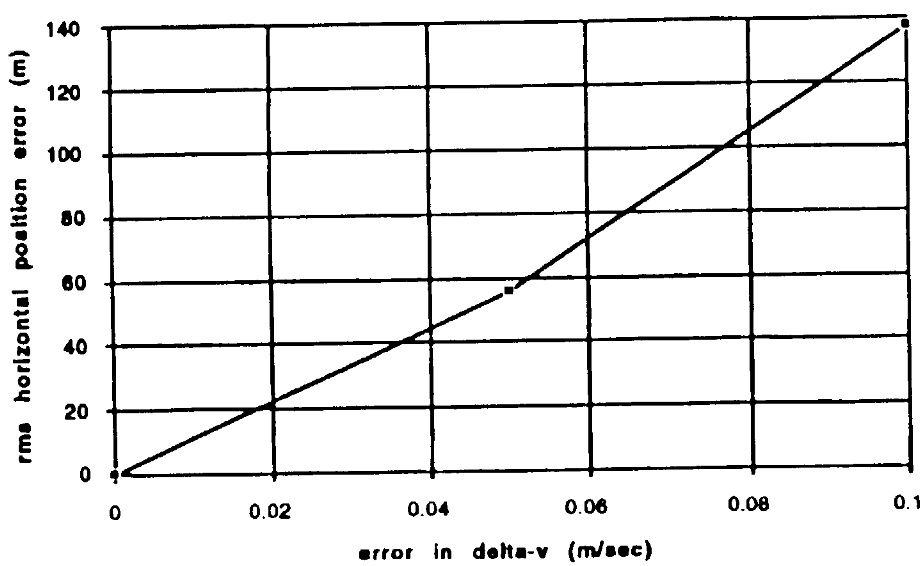






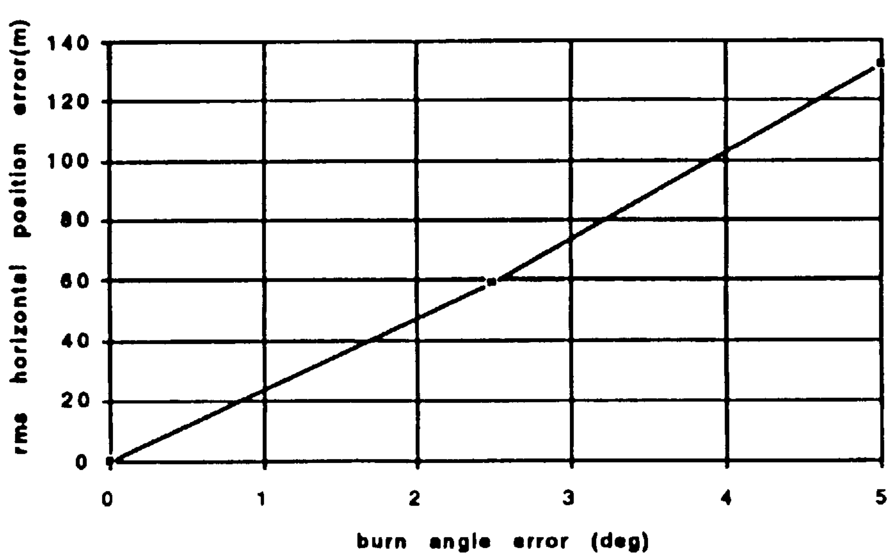
Effect of Burn ΔV Error on Landing Position

- Nominal case:
  - Asteroid mass = 10^{13} kg
  - Asteroid dimensions = 1.0 x 0.5 x 0.5 km
  - 2 mascons, 0.4 km apart
  - Rotational period = 10 hours
  - Landing approach = single impulsive burn from 1.5 x 30 km transfer orbit



Effect of Burn Angle Error on Landing Position

- **Nominal case:**
  - Asteroid mass = 10^{13} kg
  - Asteroid dimensions = 1.0 x 0.5 x 0.5 km
  - 2 mascons, 0.4 km apart
  - Rotational period = 10 hours
  - Landing approach = single impulsive burn from 1.5 x 30 km transfer orbit



Effect of Burn Angle Error on Landing Position

Error = 5° (1s)

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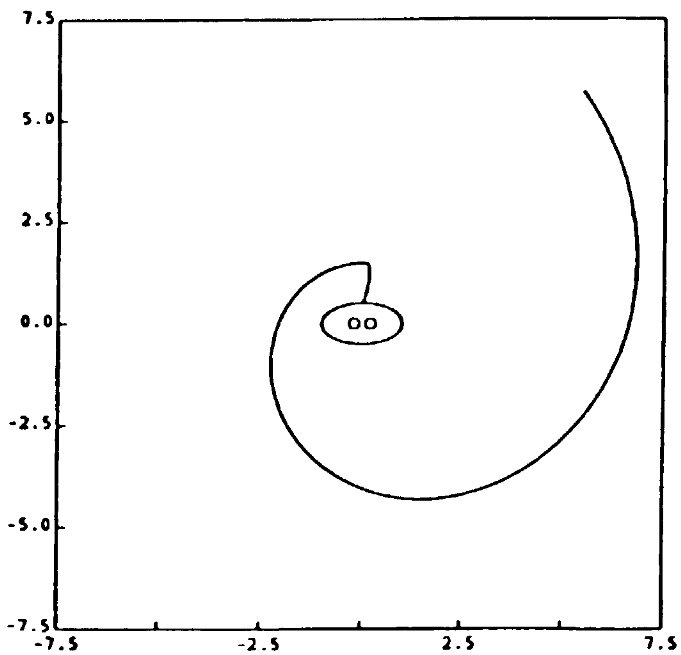
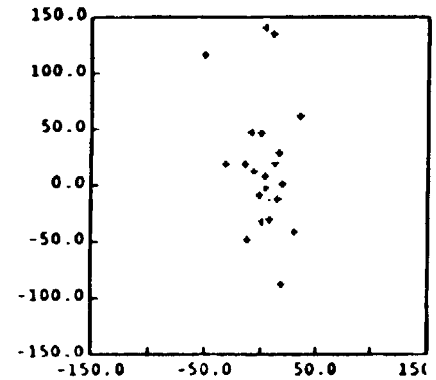
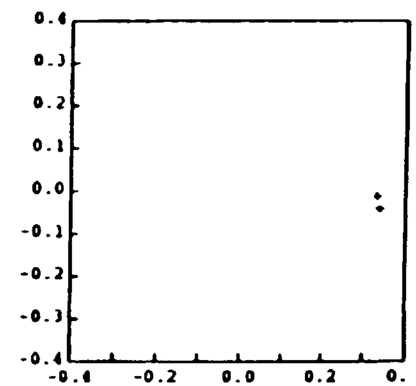






Gravity and Burn Errors

- Gravity and burn errors (for reasonable values of parameters) have very small effect on landing position
- They also have negligible effect on landing velocities
- Burn errors with sigma of 1 mm/sec and 0.1° will yield:
  - Landing position errors of 3.8 m
  - Horizontal velocity errors of 0.003 m/secand
  - Vertical velocity errors of 0.05 m/sec
- These sources of error do not prevent open loop landing

Initial State Errors

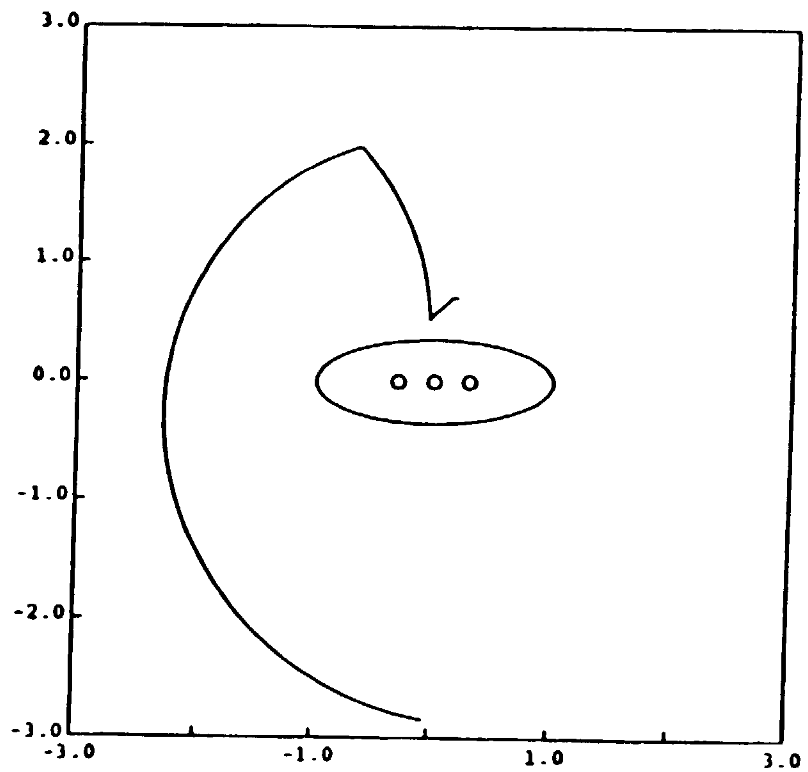
- Heritage: NEAR navigation studies
- Initial state of spacecraft can be determined to
  - ± 5 m in position
  - ± 0.005 m/sec in velocity
- What will be the effect of these errors on open loop landings?

Initial State Errors

- Results from Monte Carlo simulation of brake and puff descent, including initial state errors of 5 m position and 5 mm/sec velocity
- Vertical velocity error: 0.38 m/sec
- Landing epoch error: 560 seconds
- Possibility of "overpuff" (landing abort due to early braking and deceleration burns)

Overpuff

Example of overpuff trajectory



Initial State Errors - Results

- Initial state errors are dominant for open loop landing concept
- Open loop control can work if landing velocities of ~1 m/sec are permitted
- Lower landing velocities can be achieved with open loop control on restricted portions of the asteroid

Conclusions

- Landing should consist of four phases:
  - Initial parking orbit
  - Transfer orbit
  - Initial free fall near quasi vertical trajectory
  - Final brake and puff descent
- Spacecraft should have an altimeter and energy absorbing pads
- Nominal operation:
  - Closed loop with altimeter-triggered control
  - Landing at ~0.1 m/sec
- Fall back mode:
  - Open loop
  - Landing at 1 m/sec

Mission Operations Overview

The Mission Operations Center (MOC) will be at the Applied Physics Laboratory. Located in this center is a Mission Operations Team which is responsible for monitoring the status of the spacecraft and payload subsystems, commanding the spacecraft and payloads, and coordinating real-time mission planning with all members of other NEARS teams and participating organizations. Also in the MOC is the Mission Operations Ground System, the hardware/software system used by the Mission Operations Team.

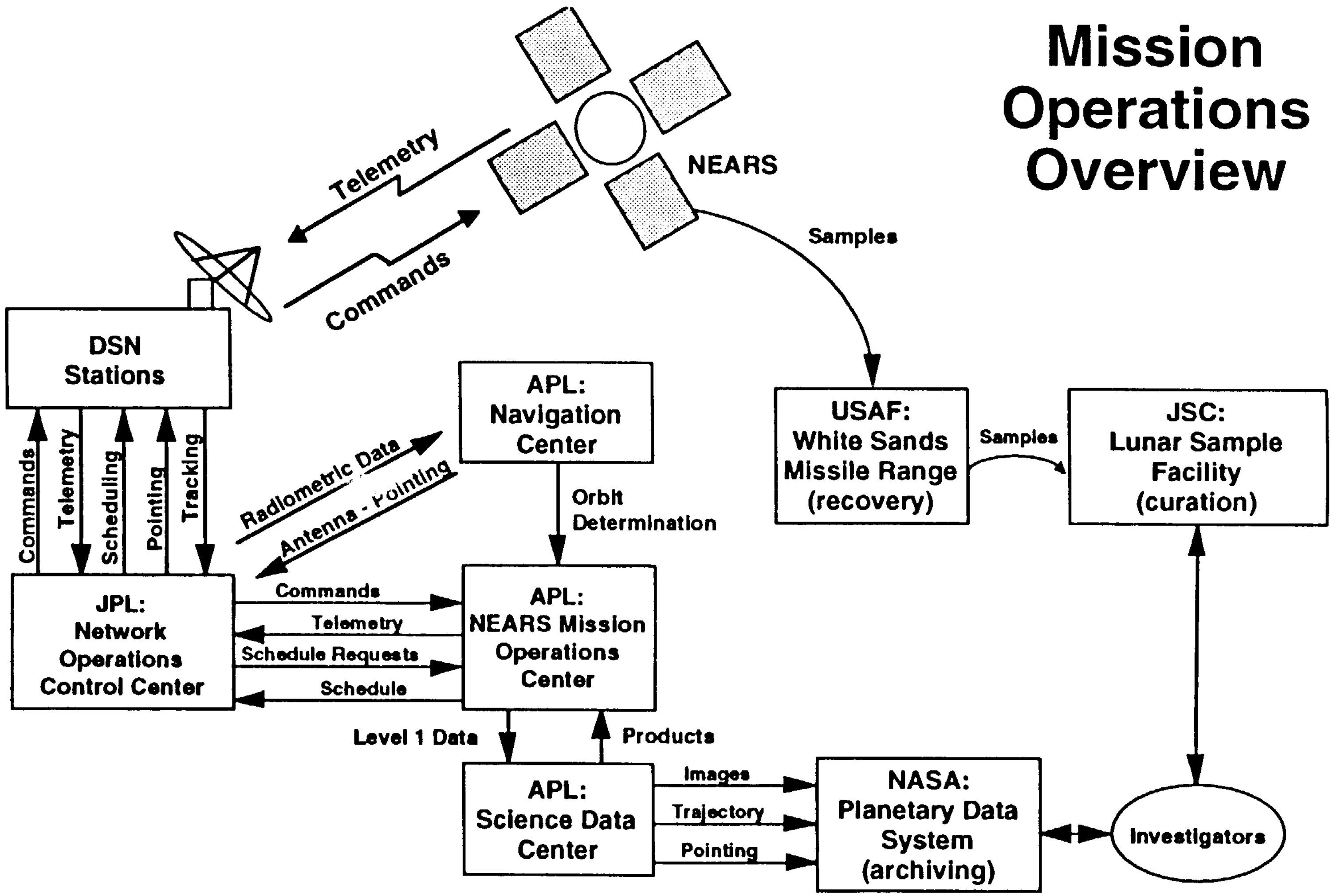
The Deep Space Network (DSN) will be used as the communications network with the Network Operations Control Center (NOCC) at JPL as the responsible facility.

We plan that NEARS navigation functions (trajectory determination, state vector production, ΔV calculation, and DSN antenna pointing angles) will be carried out by the APL Navigation Center, operating on tracking data received by the DSN and forwarded via the JPL NOCC to APL.

Imaging, ranging and trajectory data returned from NEARS will be formatted at the APL Science Data center and promptly forwarded to NASA to be archived as CDs with the Planetary Data System. Our intent is to make these data as widely available as possible.

We plan that samples will be curated at the Lunar Sample Facility of the Johnson Space Center and will be distributed according to the same protocols as lunar samples. Leading investigators from both the U.S. and abroad will be encouraged to analyze the samples. Because of the limited supply of asteroid samples, a strong Asteroid Sample Analysis Review Board will need to be established by NASA to oversee the allocation to investigators.

Mission Operations Overview



Mission Timeline (49-month Nereus Mission)

The NEARS mission has three brief periods of intense activity connected by two long, quiet cruise periods.

The mission starts a 2-week launch and early operations phase followed by a 21-month outbound cruise with no maneuver, flyby, or other major event. The cruise ends with a 2-week deceleration period, a slow flyby to the Sun side of the asteroid, and then an injection into asteroid orbit.

Operations at Nereus are limited to 68 days. The first 30 days in Nereus parking orbit are dedicated to asteroid surveys, measurements, and sample-site selection. The remaining 38 days is available for collection of samples. Each sample operation requires 3 days, including a de-orbit burn, touchdown, sample-taking, re-orbit burn, and 2 days of burns to fix up the parking orbit for the next sample.

After a 2-week Nereus departure phase, the spacecraft embarks on a 24-month return cruise, again with no maneuver, flyby or other major event. This ends with two weeks of preparations for Earth touchdown of the reentry capsule at the White Sands Missile Range.

Near Earth Asteroid Returned Sample

MARTIN MARIETTA
ASTRO SPACE

LOWELL OBSERVATORY

Mission Timeline (49-month Nereus Mission)

Launch and Early Operations (2 weeks)

Jan 10, 2000	L = Launch
L + 30 min to L+ 90 min	Transfer orbit inject
L + 90 min to L + 2 weeks	Cleanup maneuvers

Outbound Cruise (21 months)

L + 2 weeks to AN - 2 weeks	AN = Unperturbed Nereus arrival: Oct 20, 2001 Cruise (no maneuvers)

Asteroid Approach (2 weeks)

AN - 2 weeks to AN + TBD	Four ΔV burns to match asteroid velocity, inject to 15-30 km parking orbit

Asteroid Survey (30 days)

AN to AN + 30 days	Survey asteroid surface, gravity field, etc.

Asteroid Landing Phase (38 days)

AN + 30 days to AN + 68 days	Six 3-day sampling sequences

Nereus Departure (2 weeks)

Dec 27, 2001	DN = Begin departure burn
DN to DN [illegible] weeks	Cleanup burns

Return Cruise (24 months)

DN + 2 weeks to ET - 2 weeks	Cruise (no maneuvers)

Earth Return (2 weeks)

ET - 2 weeks	Orbit trim
ET - 6 hours	Release reentry capsule
ET - 5 hours	Final S/C maneuver to avoid Earth
Feb 7, 2004	ET = Earth touchdown of reentry capsule

Mission Operations Drivers

Design and development of mission operations are driven by mission characteristics that include the top-level mission timeline, spacecraft operating modes, spacecraft operating constraints, support requirements during each mission phase and event, time and information constraints on the planning process, and requirements for science data processing and dissemination.

General mission operation constraints include the 10-day launch window and the 49-month timeline with events as noted on the previous page. Because the NEARS RF system is similar to that of NEAR (except for the fan beam location), communications procedures and ground-based resources can also be similar. Selection of telemetry rates and formats is TBD, however.

Some of the more significant operations drivers for the mission phases are summarized on the folllowing pages.

Mission Operations Drivers

- Cruise Mode Constraints
  - Minimize DSN contact
  - Conserve power
  - Protect subsystems that wear out (e.g., reaction wheels)
  - No instrument operations (other than testing)
  - Ranging uplinks required every TBD days
  - Round-trip communications delay up to 50 minutes
- Asteroid Operations Constraints
  - All surveys and sampling must be completed within 70 days
  - Continuous command/telemetry coverage is needed throughout
  - Round-trip communications delay from 1.5 to 4.5 minutes
  - Maintain solar panels within 70° of full illumination
  - During Survey
    - Maintain high-gain antenna toward Earth
  - During Landing Phase
    - Point fanbeam to Earth
    - Descent to asteroid and liftoff is a single autonomous sequence

Mission Operations Drivers *(Cont.)*

- ΔV Maneuver Constraints
  - Requires spacecraft orientation plus period of continuous thrusting
  - Continuous command/telemetry coverage is needed during burn

Mission Operations Design Philosophy

Guidelines for the NEARS Mission Operations design are outlined. These are largely influenced by the approach taken for NEAR.

Existing infrastructure and facilities, such as the DSN, NASCOM, and JSCs Lunar Sample Facility will be used where cost-efficient. To maximize compatibility with the existing infrastructure, the system communications interfaces and data handling will be compliant with the recommendations of the Consultative Committee for Space Data Systems (CCSDS).

The Mission Operations Ground System architecture will be the same for spacecraft integration, test, and on-orbit operations, with use of hardware and software elements that migrate through these mission phases. Common means to test and fly the spacecraft will contribute to system reliability, save in operator training, and save in development costs.

The internal elements of the Ground System design will employ Commercial Off-the-Shelf (COTS) hardware and software products where cost-efficient. These already have a high level of operations-support capability, and this is expected to increase further with experience gained on NEAR. The use of an open operating system maximizes the availability of COTS products. Built-in networking provides flexibility in physical distribution of the system, and provides the opportunity for widespread access to mission data.

Mission Operations Design Philosophy

- **Maximize cost-efficient use of existing infrastructure:**
  - Use DSN, NASCOM
  - Maintain CCSDS Compatibility
- **Rely on NEAR Heritage:**
  - Ground System design based on NEAR
    - Open operating system (e.g., UNIX)
    - Networked architecture (e.g., EtherNet)
    - Distributed processing
    - Best use of COTS hardware and software
    - Expert-system support to operators
  - Common Ground System design for:
    - Spacecraft integration/checkout
    - Mission operations support
  - Operating methods based on NEAR
    - Small, skilled Operations Team
    - Cross-trained team members
    - Team members experienced on NEAR

Mission Operations Design Philosophy *(Cont.)*

- Operate in "surge" mode during major events:
  - Take advantage of short (70-day) duration of asteroid operations
  - Avoid cost of staffing and training large teams

Teams

NEARS activities, including mission operations, are allocated to Teams as summarized.

The Science Team consists of the investigators associated with this proposal. They are responsible for establishing mission goals and for leading the asteroid survey. They participate in instrument design,test, calibration, mission planning and mission operations at APL, data formatting and data distribution to the PDS.

The Mission Operations team is responsible for all activities required to plan, control and assess spacecraft operations. This will be a small team operating efficiently with the support of appropriate software/hardware tools. Most members routinely perform multiple tasks, and all are cross-trained to support any position. Extra manpower required for the two-month period of asteroid operations is obtained by "surge" mode operation (extended hours for the core team), and with assistance of members from other teams. During the development phase, the Mission Operations Team works with system developers to support spacecraft and ground system testing.

Teams

- Science Team
  - Led by principal investigator
  - Determines mission goals
  - Leads asteroid survey (calibrates camera, selects filter)
  - Leads landing phase (selects sites)
  - Requests Mission Operations Center to execute events
  - Participates in activities of other teams
- Mission Design Team
  - Led by Principal Investigator
  - Includes representatives of all other teams
  - Plans for asteroid operations (orbital survey and landing)
  - Plans for reentry, descent to Earth, and sample recovery
  - Designs ΔV maneuvers and provides TBD maneuver parameters to the Mission Operations Center

Teams *(Cont.)*

The Instrument Specialist Team has primary responsibility during the mission for production and dissemination of science data products (including images and trajectory information). During spacecraft development, the Instrument Specialist Team is responsible for developing the instruments.

Teams *(Cont.)*

- **Mission Operations Team**
  - Based in the Mission Operations Center
  - Obtains scheduled DSN coverage
  - Schedules APL operations
  - Commands spacecraft
  - Assesses spacecraft performance
  - Performs ΔV analysis to determine spacecraft biases and mass
  - Monitors spacecraft health
  - Produces Level 1 data from telemetry
- **Instrument Specialist Team**
  - Based in the Science Data Center during the mission
  - Establishes instrument operating procedures and guidelines
  - Produces and disseminates science data products from Level 1 data

Teams *(Cont.)*

The Navigation Team is responsible for operating the Navigation Center to provide complete navigation service for the mission. Input data consists of radiometric, imaging and altimeter data.

The remaining two teams are not active on a day-to-day basis during NEARS flight operations. However, the Spacecraft Engineering Team may be called upon for critical mission events, especially if abnormal conditions occur.

Teams (Cont.)

- Navigation Team
  - Determines state vectors from DSN ranging data
  - Produces pointing data for the DSN antennas
  - Predicts spacecraft trajectory during cruise, including effect of non-gravitational forces
  - Retrieves and interprets spacecraft attitude/orbit-related spacecraft telemetry
  - Provides asteroid orientation, spin and gravity field
  - Provides asteroid orbit prediction models
  - Monitors ΔV maneuvers and computes observed ΔV
- Spacecraft Engineering Team
  - Develops the spacecraft bus
  - Establishes operating procedures for the bus
  - Supports the Mission Operations Team for critical events
- Asteroid Sample Analysis Review Board
  - Established by NASA
  - Oversees allocation of samples to investigators

Mission Operations Functional Flow

The diagram identifies operational functions involved with NEARS, and the flow of information among those functions. This diagram is applicable primarily to the asteroid survey and landing phases of the mission.

Starting at the upper left, the Science Team using products of the Science Data Center and approved plans from the Mission Design Team, sends requests for observations and events to the Operations Planning function (at the Mission Operations Center). Visualization data (images, maps, and models) from the Science Data Center helps the Operations Planning function understand the requests. The Operations Planning function will respond to each request with the intent to generate the spacecraft command script that will carry it out. Required steps in this response are to incorporate plans from the Mission Design Team, resolve schedule conflicts, predict the effect of the proposed action/maneuver, and validate the intended script. Completed command scripts are sent to the Mission Control function for uplink to the spacecraft via the Ground Communications function (including NASCOM) and the DSN to the spacecraft.

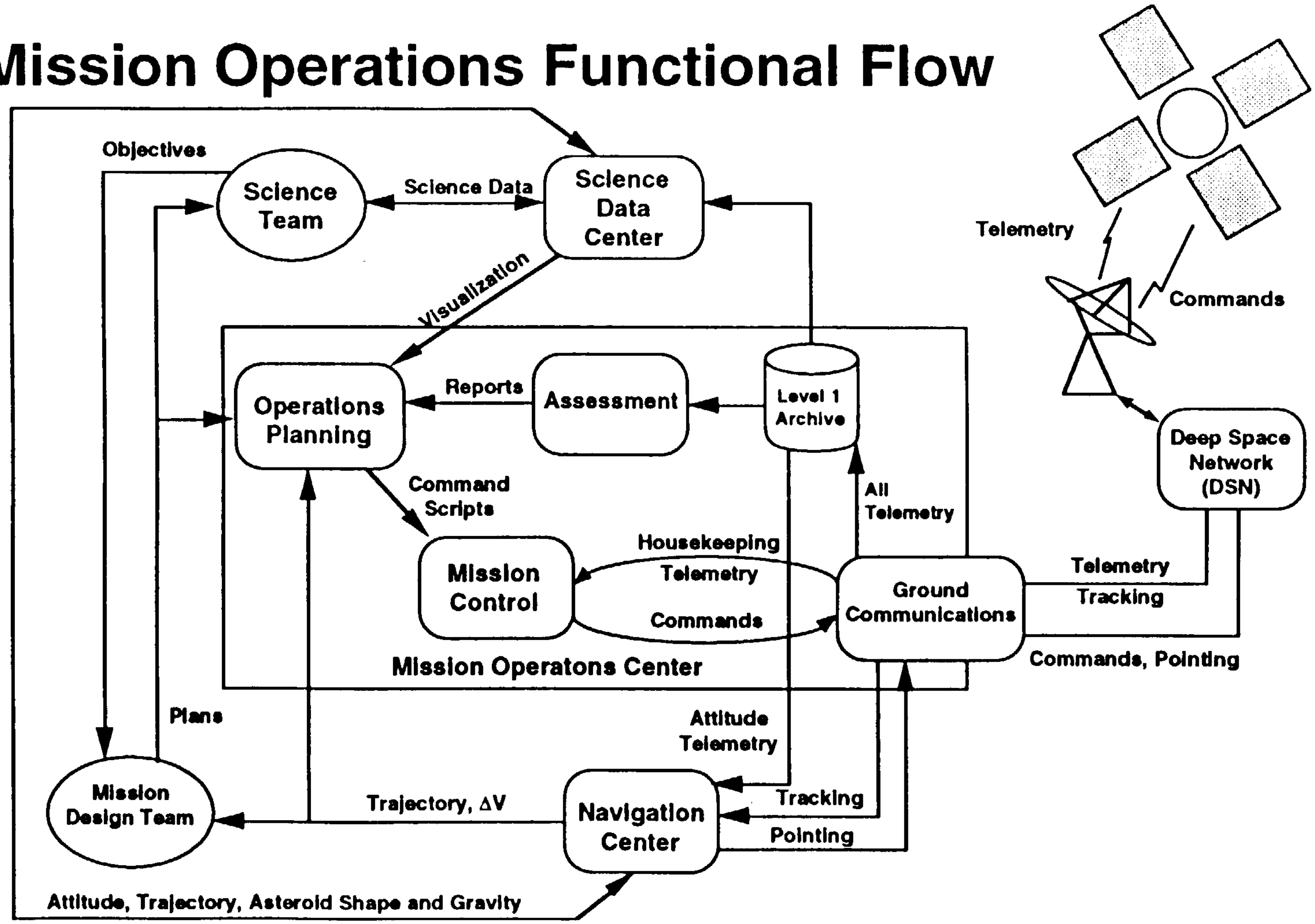
Telemetry from the spacecraft is sent via the DSN and Ground Communications function to the Mission Operations Center. For the immediate needs of command verification and spacecraft health monitoring, selected telemetry is sent directly to the Mission Control Function.

All telemetry data is sent to the Level 1 Archive to service several functions. The Assessment function uses archived telemetry and knowledge of commands determine whether intended mission operations were carried out, determine whether the spacecraft is healthy, and follow long-term trends in spacecraft status. The Science Data Center uses archived telemetry to generate images, maps, and models of the asteroid for use by both the Science Team and Operations Planning function.

At the bottom of the diagram, the Navigation Center retrieves attitude/orbit-related telemetry and receives DSN trackin g data, which it uses to provide orbital motion predictions, DSN pointing angles, and ΔV from monitored maneuvers.

The Mission Design Team combines science objectives with navigation data to generate plans for observations and events.

Mission Operations Functional Flow



Mission Activities by Phase

During launch and early operations, the Mission Operations Center is responsible for a full spacecraft systems checkout upon completion of the separation sequence. After this, models used to predict spacecraft behavior (thermal, power system, attitude, and propulsion) must be validated.

During the outbound and return cruises, one 4-hour DSN contact per week, requiring both uplink and downlink, will be conducted. On each contact, the Mission Operations Center will update the spacecraft-stored ephemeris and command memory, and monitor spacecraft health and status. The DSN will carry out ranging (and radiometric) measurements.

Routine operations for other mission phases are outlined. Contingency plans will be in place to be executed if an anomaly or spacecraft failure occurs. This is especially important during early operations, but it applies throughout the mission.

Mission Activities by Phase

- Launch and Early Operations
  - Establish initial trajectory
  - Check out spacecraft bus, instruments and payload
  - Validate models for spacecraft behavior
  - Establish cruise mode configuration
- Cruise (Outbound and Return)
  - 4-hour DSN contact once/week
- Planned ΔV Maneuvers (Asteroid Approach, Departure Burns, Orbit Trim)
  - Mission Design Team determines maneuver parameters at Burn - 1 day
  - Mission Operations Team
    - Uploads burn commands
    - Monitors burn in real time

Mission Activities by Phase *(Cont.)*

- Asteroid Survey Operations
  - Continuous DSN coverage required
  - Survey plan established by Mission Design Team and Science Team
  - Mission Operations Team plans/conducts daily operations:
    - Command uploads
    - Monitor spacecraft health/status
    - Assess performance of events
- Asteroid Sampling Operations
  - Continuous DSN coverage required
  - Sample sites selected by Science Team by arrival + 30 days
  - Mission Operations Team
    - Commands and monitors sampling events
    - Conducts daily operations, including fix-up burns, between sampling events

DSN Coverage Summary

The Deep Space Network (DSN) coverage that the Mission Operations Team proposes to request.

DSN Coverage Summary

Mission Phase	Antenna	Coverage
Transfer orbit Injection and early cruise	34 m	Continuous for 30 days
Cleanup maneuvers (M)	34 m	As needed, M ± 12 hours for 2 weeks
Outbound cruise	34 m	One 4-hour pass/week
Asteroid approach	34 m	One 8-hour pass/day for 2 weeks
Operations near asteroid	34 m	Continuous for 70 days
Asteroid touchdowns (T)	34 m	Up to six events, T ± 15 hours at 3-day intervals or as needed.
Asteroid departure & cleanup burns	34 m	As needed, M ± 12 hours for 2 weeks
Return cruise	34 m	One 4-hour pass/week
Final orbit trim maneuver	34 m	Continuous OT ± 12 hours
End of return cruise	34 m	Continuous, OT - 5 days to Earth flyby + 12 hours
Unplanned maneuvers & critical events	70 m	As needed

New Technology

- **New Technology Infusion**
  - Use of BMDO sensors in 4-year deep-space mission
  - Optional propulsion schemes use advanced thrusters in interplanetary application
  - Low-cost land recovery of reentry vehicle
- **New Technology Transfer**
  - NEARS is a pathfinder for future utilization of space resources
  - Robotic mining operations in space
  - Low-cost return of extraterrestrial materials to Earth

Education Initiatives

- Broad dissemination of asteroid images in near real time on Internet
- Provide opportunities to follow landing operations via on-line distribution of descent imaging sequences in near real time
- Involvement of university faculty and students at undergraduate and graduate levels in analysis of asteroid samples and images

Management Structure

- Management structure will be based on that for the NEAR Program: a small Headquarters Program Office works with the APL Project Office
- The Principal Investigator will be E. M. Shoemaker of Lowell Observatory, who will be responsible for the mission science return and will guide, direct and coordinate the activities of the Science Team
- The Project Manager will be T. B. Coughlin of JHU/APL, who will be responsible for the development and integration of the spacecraft, instruments, and mission operations
- The Spacecraft and Technical Lead will be A. F. Cheng of JHU/APL, who will serve as a technical lead for design of the spacecraft, asteroid operations, and sample collection system
- T. B. Coughlin and A. F. Cheng are the Project Manager and the Project Scientist, respectively, for the NEAR mission

Institutional Roles

Proposing Consortium: Institutions and Roles

Lowell Observatory

E. Shoemaker, P. I.

- Lead Science Team activities

JHU/APL

T. Coughlin, Project Manager

- Build spacecraft
- Procure/develop instruments, including sample collection system
- Perform spacecraft and sub-system test and integration
- Perform mission operations

Martin Marietta Aerospace

- Build Earth return capsule

Involvement of NASA Centers

NASA HQ

- Program-level management
- Funding of spacecraft, sub-systems, and instrument development
- Launch procurement

Jet Propulsion Laboratory

- Deep Space Network
- Planetary Data System

NASA/Johnson Space Center

- Office of the Curator

Roles of Science Team Members

- Assist in design, test, and analysis of sample collection system
- Assist in design, test, calibration, and integration of instruments
- Assist in mission design
- Assist in ground system design and development
- Serve as a resource on asteroid bulk properties and surface properties, especially roughness, hardness, cohesiveness
- Participate in planning and executing a ground-based observing campaign
- Assist in design and planning of asteroid operations, including survey and landing phases

Roles of Science Team Members *(Cont.)*

- Plan and perform orbital survey of asteroid target
- Prepare orbital data for archiving in PDS
- Assist in selection and characterization of landing sites
- Plan and perform laboratory analyses of asteroid samples
- Participate in distribution and curation of samples
- Analyze geologic context of asteroid samples
- Assess relationships between asteroidal samples and meteoritic, cometary, terrestrial, and lunar materials

New Start Dates

Prime mission opportunity is January 2000 launch to Nereus

New start can be in January 1997 (FY 97) for 36-month program

Near Earth Asteroid Returned Sample

MARTIN MARIETTA
ASTRO SPACE

LOWELL OBSERVATORY

Preliminary Schedule